NACA

RESEARCH MEMORANDUM

A PRELIMINARY INVESTIGATION OF METHODS FOR IMPROVING

THE PRESSURE-RECOVERY CHARACTERISTICS OF VARIABLE-

GEOMETRY SUPERSONIC-SUBSONIC DIFFUSER SYSTEMS

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UNCLASSIFIED

To _____

NACA Res ahd
/RN-126
By authority of _____ Date _____ Apr. 15, 1958

AMT 5-13-58

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS
WASHINGTON
October 16, 1957

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A PRELIMINARY INVESTIGATION OF METHODS FOR IMPROVING

THE PRESSURE-RECOVERY CHARACTERISTICS OF VARIABLE-

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SUMMARY

An investigation has been initiated to study methods for improving
the pressure-recovery characteristics of two-dimensional, variable-
geometry, supersonic-subsonic diffuser systems. The recovery character-
istics of the basic configuration and of a configuration with an injector
at the end of the supersonic diffuser have been determined at Mach num-
bers from 2.5 to 4.75, and at stagnation pressures from 40 to 205 pounds
per square inch absolute. The Reynolds number based on the height of
the test section and a stagnation pressure of 120 pounds per square inch
absolute varied from 12.2×10^6 to 4.0×10^6 over the Mach number range
from 2.5 to 4.75.

The pressure recovery of the basic configuration, which had a rela-
tively long supersonic diffuser to minimize shock boundary-layer inter-
action effects, varied from 0.71 to 0.21 as the test-section Mach number
varied from 2.5 to 4.75 for a stagnation pressure of 120 pounds per square
inch absolute. These recoveries are significantly higher than the corre-
sponding theoretical normal-shock recoveries which vary from 0.50 to 0.075.
For a given contraction ratio of the supersonic diffuser, the effect of
Reynolds number on pressure recovery was negligible. The maximum recovery
decreased somewhat as the Reynolds number decreased because the amount
of contraction of the supersonic diffuser decreased. Use of a Mach num-
ber 2.19 injector with a relatively high mass flow (injector to main-
stream mass-flow ratio of 2.5 at a test-section Mach number of 4.0) at
the end of the supersonic injector resulted in recoveries which varied
from 0.65 to 0.56 as the test-section Mach number varied from 2.95 to 4.70.

The results indicate that the maximum test Mach number of existing
supersonic wind tunnels with limited compression ratio and conventional
normal-shock diffusers can be significantly increased by modifying the
diffuser to incorporate a relatively long variable-geometry supersonic
diffuser followed by an injector. For example, a compression ratio
of 2.5 instead of about 20 is sufficient for a test Mach number of
about 5. The power consumption of high Mach number tunnels with conven-
tional normal-shock diffusers can be significantly reduced by modifying
the tunnel to incorporate a relatively long variable-geometry supersonic
diffuser.

INTRODUCTION

The rapid increase in the supersonic performance of airplanes and
missiles has generated major technical problems in the design of the
companion wind-tunnel research facilities. In less than a decade the
operational speeds of airplanes have increased from near sonic to a Mach
number of 3, while some missiles are already operating at a Mach number
of 5 and above. This upsurge in flight speeds has made the Mach number
range of existing wind-tunnel facilities inadequate.

At Mach numbers to about 5 the maximum test Mach number is gener-
ally determined by the compression ratio of the drive equipment. For
conventional wind tunnels where the normal shock occurs essentially at
test-section Mach number, a compression ratio of 2 is required for a
Mach number of 2. A Mach number of 5, however, requires a compression
ratio of 20. A continuous-operation facility for this Mach number range
requires an elaborate compressor system with complex staging. There is
a great need for research on methods for decreasing the compression-
ratio requirements of supersonic wind tunnels, not only from the stand-
point of simplifying the design of new facilities but also for improving
the supersonic performance of those now in operation.

Numerous studies have been made of methods for improving the recov-
ery characteristics of wind tunnels (refs. 1 to 8, for example). Most
of these results are summarized in reference 1. The best results were
obtained by the use of a variable-geometry supersonic diffuser down-
stream of the test section to reduce the Mach number at which the nor-
mal shock occurred. Further improvements might be obtained by the use
of some form of boundary-layer control such as that used with inlets
(refs. 9 to 11) since the measured recoveries of the wind-tunnel con-
figurations are well below calculated values.

Existing low Mach number tunnels have relatively constant-volume
compressors and constant-area test sections. If these tunnels are to
be operated at higher Mach numbers a considerable amount of air must
bypass the test section in order to satisfy the compressor flow require-
ments. Consideration may therefore be given to methods of increasing
wind-tunnel recoveries by reintroducing into the tunnel the bypassed
air by means of injectors. The effects of air injection immediately
downstream of the test section have been considered in references 7
and 8. The resultant recoveries were not as good, however, as those
obtained from the variable-geometry diffuser tunnels.

The combination of a variable-geometry supersonic diffuser and a
supersonic injector has not been studied previously. Location of the
injector at the end of the supersonic diffuser might improve the pres-
sure recovery by several favorable effects. First, the injector air

may have a higher total pressure at the injection point than the main airstream since the latter has encountered viscous losses and the oblique-shock-wave losses in the supersonic diffuser. The injector air may therefore mix with main airstream and raise the overall total pressure. Second, the injector air may be introduced into the tunnel at a static pressure which is higher than the static pressure of the main airstream and thereby further decelerate the latter. Third, the injection air may serve as a means of boundary-layer control by reenergizing the boundary layer at the end of the supersonic diffuser and thereby reduce the losses due to the normal-shock—boundary-layer interaction.

It thus appears that several methods are available whereby the recoveries of supersonic wind-tunnel diffusers may be further improved. The effectiveness of these methods can only be determined by experimental studies. A two-dimensional, variable-geometry, supersonic-diffuser apparatus has therefore been built to study various methods of improving the pressure-recovery characteristics of supersonic wind tunnels. This apparatus has been designed so that the effects of supersonic-diffuser configuration and contraction ratio, boundary-layer control, air injection, Mach number, Reynolds number, and a typical research model configuration can be evaluated. Inasmuch as the problems of pressure recovery in supersonic inlets are similar to those of wind tunnels, the results of this investigation may assist in the design of high Mach number inlets.

This report presents the results obtained from the basic supersonic-diffuser configuration and some preliminary results obtained by air injection at the end of the supersonic diffuser. The Mach number was varied from 2.5 to 4.75. The Reynolds numbers corresponded to the values for a tunnel with a 4.5-foot, square test section and with stagnation pressures of 0.3 to 1.7 atmospheres.

SYMBOLS

A   area, sq ft

w   air flow, lb/sec

M   Mach number

p   static pressure, lb/sq ft

p_t total pressure, lb/sq ft

δ wall angle or flow turning angle, deg

Subscripts:

∞ free stream or test section

a   atmospheric

j   injector

ts   test section

av   mass-flow weighted average pressure

b,c,d,e,f, diffuser-apparatus station (see figs. 1 and 2)
  and so forth

bc, ef   wall section of diffuser apparatus

DESIGN AND DESCRIPTION OF APPARATUS

General

The diffuser apparatus is shown schematically and pictorially in
figures 1 to 4. The apparatus consisted of fixed side walls which were
6.75 inches apart and movable top and bottom walls. Each movable wall
(figs. 1 and 2) was made up of 7 hinged sections. The names assigned
to the various sections in figure 1 describe in general terms their
respective functions when the injector was located at station d. In
figure 1 all wall sections except the supersonic nozzle and subsonic
diffuser have been drawn parallel to the center line to show their
actual lengths. Typical wall settings which were used during the inves-
tigation are shown in figure 2 for configurations with and without the
main injector. The assembled apparatus is shown in figure 3. The photo-
graphs of figure 4 were taken with one side wall removed to illustrate
details of the various parts of the apparatus. The various wall sections
were in random positions when the photographs were taken and these posi-
tions do not represent experimental wall settings.

Each section of the apparatus is described in detail in the
following paragraphs of the text. Included in the description is a
discussion of the basic considerations involved in the design of each
section. This discussion has been included to describe the aerodynamic
function of each section of the apparatus and to describe the overall
design philosophy.

Supersonic Nozzle

The variable Mach number flow was generated by the nozzle shown in figures 1, 2, and 4(a). The nozzle was formed by fixed contour blocks which pivoted about fixed hinges at station 24. The nozzle was designed to produce a uniform flow at station 24 at a Mach number of 3.3. Other test Mach numbers from 2.5 to 4.75 were obtained by pivoting the blocks to change the height of the nozzle throat. It was recognized that this method of varying the Mach number would produce a nonuniform flow at station 24 at off-design Mach numbers. This method was chosen, nevertheless, because of its mechanical simplicity and because it was thought that some flow variation at station 24 would not seriously affect the accuracy of the results of the investigation.

Main Supersonic Diffuser

The supersonic diffuser (figs. 1, 2, and 4(a)) is a very important part of any diffuser apparatus designed for high pressure recovery. The purpose of the supersonic diffuser is to efficiently decelerate the air from the initial Mach number to a minimum supersonic Mach number at the normal shock. Theoretically the deceleration could be accomplished to a Mach number of 1 with no loss of pressure recovery in either a reverse supersonic nozzle or a very long, straight-walled, decreasing-area duct. Practically, of course, this ideal deceleration cannot be attained.

For wind-tunnel installations a straight-walled, variable-area, supersonic diffuser of minimum length is probably most suitable because of its mechanical simplicity. Several factors must be considered when selecting the diffuser length. In a diffuser of this type the deceleration of the air is accomplished by a series of oblique shock waves which reflect several times from the diffuser walls. The turning angle of these waves equals the diffuser wall angle. As the diffuser becomes shorter the strength of each shock wave must become stronger to accomplish the same overall Mach number reduction. Theoretical calculations indicate that as the shock waves become stronger the associated total-pressure losses increase. These losses become relatively large when the turning angle of each shock wave exceeds 5° or 6°.

As the diffuser becomes shorter and the wall angle increases to maintain the same area reduction, the possibility of creating boundary-layer separation increases because the static-pressure rise at the points of shock-wave reflection becomes larger. The pressure rise required to separate a turbulent boundary layer under these conditions has been studied experimentally (ref. 12, for example). The tests of reference 12 were conducted on a flat plate with a fully turbulent boundary layer and with a Reynolds number, based on momentum thickness, which exceeded several thousand. The results of this investigation are presented in

figure 5, and indicate that the boundary layer on a flat plate may separate when the turning angle of the reflected oblique shock wave exceeds about 8°. Since the boundary layer must traverse several shock reflections in the straight-wall supersonic diffuser, it would seem advisable to limit the shock-wave turning angle to 5° or 6° and thus minimize the possibility of boundary-layer separation.

With the preceding considerations in mind, a diffuser was designed with a length—test-section-height ratio of 4.37. This diffuser when set with a wall angle of 5° would decelerate the flow from a Mach number of 4.0 to a Mach number of 2.45 at station c. The top and bottom walls of the main supersonic diffuser pivoted about the fixed hinge at station 24 and incorporated a movable hinge and a sliding joint at the downstream end (figs. 1 and 4(a)). The relationship between the diffuser wall angle δ_{bc} and the diffuser-contraction ratio A_c/A_b is presented in figure 6. The Mach number at the end of the diffuser M_c has been computed for various values of initial Mach number M_∞ and contraction ratio A_c/A_b by the use of the oblique-shock-wave equations, neglecting viscous effects. These results are presented in figure 7.

Injector

The use of injectors after the test section as a means of increasing the diffuser pressure recovery has been discussed in references 7 and 8. No mention was made, however, of the advantages which may result from the combined use of a variable-geometry diffuser and an injector. Location of the injector at the end of the supersonic diffuser permits the injector air to enter the diffuser at a relatively low Mach number when compared to the test-section Mach number and at a total pressure which is higher than that of the main airstream since the latter has encountered viscous losses and oblique-shock-wave losses in the supersonic diffuser. The injector air may therefore mix with the main airstream and raise the average total pressure ahead of the normal shock. Location of the injector at the end of the supersonic diffuser may also be a method of boundary-layer control. The injected air may reenergize the boundary layer which has formed in the supersonic diffuser and thereby reduce the normal-shock boundary-layer interaction losses. Finally, as was pointed out in references 7 and 8, the injector air may be introduced into the tunnel at a static pressure which is higher than that of the main air stream and thereby further decelerate the latter. In this process the injector air is necessarily accelerated to some extent.

The injector installation is shown in figures 1, 2(a), and 4(b). The main injector section was designed to move in a vertical direction only. As a result the diffuser wall from stations 53.5 to 61.83 was

always parallel to the tunnel center line. The air was injected at station 61.83 through a nozzle whose axis was inclined 10° to provide more strength for the injector nozzle. The nozzle contours were designed for a Mach number of 2.0. The nozzle width diverged to some extent downstream of the throat for structural reasons and as a result, the injector Mach number based on the nozzle exit to throat areas was 2.19. The injector Mach number should be variable to obtain maximum pressure recovery over the test-section Mach number range, but for this apparatus such an arrangement was not practical. The injector Mach number was chosen so that over the anticipated range of test conditions the static-pressure differences between the merging streams would be as small as possible. Thus, the strengths of the resultant shock waves would also be kept to a minimum and the possibility of initiating boundary-layer separation lessened.

If the injector were to serve as a scheme for matching the air flow of the compressor and test section, the injector mass flow should vary with test-section Mach number. Such a design was not practical for the present apparatus. Therefore the constant mass-flow injector was designed to provide the necessary matching characteristics at a relatively low test-section Mach number. The ratio of the injector to the test-section air w_j/w_{ts} for the diffuser apparatus is presented as a function of test-section Mach number in figure 8. This ratio varied from 1 to 4.8 as the Mach number varied from 3.0 to 4.75. For comparison purposes the injector mass-flow ratio which would be available at various Mach numbers for a typical low-compression tunnel is also presented in figure 8. The tunnel was assumed to have a fixed test-section area and a constant volume compressor of the correct capacity for a Mach number of 2. For this tunnel, sufficient bypass air would be available for the injector used in this investigation at Mach numbers above 2.6.

Mixing Section

The mixing section (figs. 1, 2, 4(b), and 4(c)) was incorporated in the diffuser apparatus to promote mixing of the injector and main airstream before encountering the normal shock. The length of the section was chosen in a somewhat arbitrary manner and was thought to be too short for adequate mixing at large values of supersonic-diffuser contraction ratio A_c/A_b. This wall section was parallel to the horizontal center line for all of the injector tests.

During the no-injector tests the upstream end of the mixing section was flush with the main injector surface as shown in figures 2(b) and 4(c). For these tests the wall was set at an angle of 1° to produce a slightly divergent section.

Second Supersonic Diffuser

The second supersonic-diffuser section was added to provide addi-
tional compression after the mixing section. The boundary layer which
developed on the injector nozzle and mixing section surfaces was thought
to be able to withstand some adverse-pressure gradient without separation.
Use of the diffuser section should therefore increase the overall pres-
sure recovery of the apparatus. Overall length considerations of the
apparatus required that this section must be relatively short. The con-
traction ratio A_f/A_e of this section is a function of the contraction

ratio A_c/A_b and the wall angle δ_{ef}. This relationship is given in
figure 9. During the no-injector tests the wall angle of this section
was set at 1° (fig. 2) to continue the diverging area passage of the
mixing section.

Normal-Shock Section

The normal-shock section was designed to reduce the total pressure
losses across the normal shock. Several investigations (refs. 13 to 15,
for example) have shown that the deceleration from supersonic to sub-
sonic speeds usually occurs through a series of shocks instead of a sin-
gle shock when the pressure rise across the shock is sufficient to sepa-
rate the boundary layer. These results also show that the complete shock-
wave pattern should occur in a passage of nearly constant area if the
normal-shock boundary-layer interaction losses are to be minimized. On
the basis of this information the normal-shock section of the diffuser
apparatus (figs. 1 and 2) was made 19 inches long. This length was some-
what arbitrary and was thought to be too short for large values of dif-
fuser wall height at station f. As shown in figure 1 the normal-shock
section included a sliding joint. The portion of the wall upstream of
the joint had a wall angle of 2°; the downstream section of the wall was
designed to be parallel to the tunnel center line.

Subsonic Diffuser

The subsonic-diffuser design (figs. 1, 2, and 4(d)) was dictated
by dimensional requirements of the overall apparatus instead of the
conventional requirements for efficient subsonic deceleration of the
air. As a result the diffuser wall angles for running conditions were
relatively high (fig. 2). A splitter plate was placed on the tunnel
center line for the entire length of the subsonic diffuser in an attempt
to improve the diffusion. The actual effectiveness of such a fix is
not known. It must be remembered that for this apparatus and for all
high Mach number wind-tunnel diffusers designed for high recoveries the
subsonic-diffuser performance would have only a minor effect on the

overall pressure recovery. If the overall length is limited, design
compromises should be made in the subsonic-diffuser section rather than
in the supersonic-diffuser section.

Downstream End of Apparatus

The downstream end of the apparatus extended from the end of the
subsonic diffuser, station 116.25, to station 159 (figs. 1, 2, and 4(d)).
This portion of the apparatus was designed to facilitate the experimental
tests and would not be included in an actual wind-tunnel diffuser. The
section from stations h to k was designed to provide a more nearly uni-
form distribution of total pressure at the rake station k. The exit total
pressure was measured by a 15-tube rake located on the vertical center
line of the apparatus at station k. Ten of the tubes, equally spaced
from top to bottom, were manifolded to a small pressure-averaging cham-
ber. The exit total pressure was assumed to be equal to the pressure
in this chamber. The other 5 tubes, also equally spaced, were used to
check the total-pressure distribution at station k. The butterfly valve
was used to control the exit back pressure.

Typical Research Model

The research model, including its support system, which is usually
located in the test section of a wind tunnel may have an adverse effect
on diffuser pressure recovery. These effects were studied by use of a
typical research model (fig. 10) which consisted of a body of revolution
and sting at an angle of attack of 15°, and the associated support strut.
All of the components were approximately scaled copies of the equipment
used in the Langley 4- by 4-foot supersonic pressure tunnel. The leading
edge of the strut was located at station 24.0.

TESTS

General

The diffuser apparatus was installed in a test cell of the Langley
gas dynamics laboratory and was operated with dry air from the high-
pressure air-storage facilities. Most of the tests were run at stagna-
tion pressures of 40 to 205 pounds per square inch absolute and at an
average stagnation temperature of about 130° F. The Reynolds number,
based on a test-section height of 6.75 inches and a stagnation pressure
of 120 pounds per square inch absolute, varied from 12.2×10^{6} at

$M_\infty = 2.5$ to 4.0×10^6 at $M_\infty = 4.75$. These Reynolds numbers correspond
to the values for a tunnel with a test section which is about 4.5 feet
square and with a stagnation pressure of 1 atmosphere.

The pressure recovery was based on the exit total pressure at sta-
tion k. This exit pressure was assumed to be the average pressure
obtained by manifolding to a common chamber the pressures of 10 equally
spaced total-pressure tubes located on the vertical center line of the
apparatus.

The pressure data were measured with Bourdon-tube pressure gages
or mercury manometer boards. These data and the wall positions, which
were indicated by the counters visible in figure 3, were photographically
recorded.

The diffuser airflow characteristics were continuously observed
during the tests by means of a shadowgraph system. These shadowgraphs
were photographed simultaneously with the pressure data.

Procedure

The pressure-recovery data were obtained in the following manner.
After supersonic flow was established in the diffuser the movable walls
were set in the desired positions. Then the back pressure was gradually
raised by closing the butterfly valve until the flow in the main super-
sonic diffuser became subsonic. The average exit total pressure at sta-
tion k was read visually on a pressure gage at the instant of flow break-
down and this pressure was used in calculating the pressure recovery.

During the tests, wall sections cd, de, ef (no-injector configura-
tion only), and fg were maintained in the angular positions shown in
figure 2. Preliminary tests with no injector indicated that the pres-
sure recovery was not improved by increasing the angle of subsonic-
diffuser wall sections de and ef to 3°.

All the starting data and minimum running-contraction-ratio data
were obtained with the butterfly valve completely open and with the top
and bottom wall angles beyond the minimum area station approximately as
shown in figure 2(b).

Accuracy

The accuracy of the pressure-recovery data cannot be evaluated in
specific numbers, but it is thought to be good on the basis of the facts
discussed below, and on the basis of data repeatability which was about

±0.01 to ±0.02, depending on the stagnation pressure. The accuracy is primarily dependent upon the uniformity of the total-pressure distribution at the rake station k. As previously mentioned in this report, the apparatus from stations h to k was designed to promote a uniform distribution at the rake. The uniformity in the vertical plane of the rake was checked by 5 total-pressure tubes equally spaced from top to bottom and was found to be about ±0.01 of the stagnation pressure. The numerically averaged pressure of these 5 tubes agreed with the pressure obtained from the 10 manifolded total-pressure tubes. No checks were made of the horizontal distribution. However, it would seem reasonable that this distribution should be as good as the vertical distribution.

The Mach number distribution at the end of the supersonic nozzle was not uniform (fig. 11) at the higher test-section Mach numbers because of the nozzle design. These distributions were obtained by measuring the static pressures on one side wall at station 23.7 and computing the Mach number from the static to stagnation pressure ratio. The test-section Mach number M_∞ was obtained by averaging these distributions. Since the survey did not extend to the top and bottom walls the resultant accuracy of M_∞ is probably no better than ±0.1 at the higher Mach numbers. An error of this magnitude was considered acceptable for the type of investigation being made.

Corrections

The total pressure of the injector air was about 10 percent less than the corresponding pressure $p_{t,\infty}$ of the test-section air because of piping losses. These losses have been taken into account by use of an average total pressure $p_{t,av}$ as the reference pressure for all data obtained with the injector configuration. This pressure is the mass-flow-weighted average total pressure of the injector and test-section airstreams. The maximum correction occurred at a Mach number of 4.70 where the average total pressure was about 0.92 of the test-section total pressure. Use of $p_{t,av}$ instead of $p_{t,\infty}$ as the reference pressure raised the maximum pressure recovery at a Mach number of 4.70 from a measured value of 0.52 to a corrected value of 0.56.

RESULTS AND DISCUSSION

Without Injection

Pressure recovery.- The basic pressure-recovery results which were obtained at Mach numbers of 2.50 to 4.75 from the diffuser apparatus

with no injector at station 61.8 are presented in figure 12. The meas-
ured recovery is plotted as a function of the contraction ratio of the
main supersonic diffuser A_c/A_b. The computed recovery accounts for the
oblique shock losses in the main supersonic diffuser but neglects the
effects of viscosity. The normal shock was assumed to occur at a Mach
number equal to M_c. The measured pressure recoveries (fig. 12) increase
at a given Mach number with decreasing values of contraction ratio of
the supersonic diffuser A_c/A_b as would be expected. The rate of
increase is less than predicted by the computed recoveries, probably
because the viscous losses due to normal-shock—boundary-layer inter-
action increase as the adverse pressure gradient in the supersonic dif-
fuser becomes larger. The pressure recoveries at a given contraction
ratio were not significantly affected by Reynolds number (varying stag-
nation pressure) and were reduced slightly when the model was placed in
the airstream.

The maximum measured pressure recoveries and the corresponding
minimum running-contraction ratios of the supersonic diffuser A_c/A_b
are summarized in figures 13 and 14. The computed recovery in figure 13
is based on the contraction ratio at which the maximum recovery was
measured at 120 pounds per square inch absolute.

The maximum measured pressure recovery (fig. 13) at a stagnation
pressure of 120 pounds per square inch absolute varied from 0.71 at
$M_\infty = 2.50$ to 0.24 at $M_\infty = 4.50$. (This stagnation pressure corresponds
to the Reynolds number for a tunnel with a 4.5-foot square test section
and with atmospheric stagnation pressure). These maximum recoveries are
significantly higher than the corresponding test-section normal-shock
recoveries which vary from 0.50 to 0.075, but are from 0.15 to 0.24
lower than the computed recoveries. Most of the computed loss of total
pressure is due to the normal-shock loss. The losses due to the oblique
shock system in the supersonic diffuser are a maximum at $M_\infty = 4.50$ and
are equal to about 0.07 of the stagnation pressure. At $M_\infty = 2.5$ the
oblique shock losses are negligible.

The maximum recoveries decreased as the stagnation pressure
decreased because the minimum running-contraction ratios (fig. 14) became
larger. This is probably a Reynolds number effect. The addition of a
model reduced the maximum recoveries at a stagnation pressure of
120 pounds per square inch absolute by 0.05 to 0.10. This loss in pres-
sure recovery was also primarily caused by an increase in the minimum
running-contraction ratios (fig. 14).

The variation of the minimum running-contraction ratio with Mach
number (fig. 14) is similar to that required to isentropically deceler-
ate the Mach number to 1.0 at station c. The minimum values at a

stagnation pressure of 120 pounds per square inch absolute are about 0.1 larger than the theoretical minimum contraction ratios. This difference increased when the Reynolds number was decreased and when a model was placed in the airstream. The factors which may determine the minimum running-contraction ratio are discussed in a subsequent section of this report.

In principle the diffuser apparatus is similar to an internal-contraction supersonic inlet, and the comparison of the respective recoveries is presented in figure 15. Two of the inlets (refs. 16 and 17) were two-dimensional designs with one movable straight wall. The supersonic diffuser of the third two-dimensional inlet (ref. 18) was shaped like a reverse supersonic nozzle. The other inlet (ref. 19) was axisymmetric with a movable, conical centerbody. The maximum recoveries of the diffuser apparatus are 0.06 to 0.08 less than the best recoveries obtained from the internal compression inlets. Two factors should be considered in the comparison of these recoveries. First, the recoveries of the diffuser apparatus would probably have been improved to some extent if the boundary layer generated by the supersonic nozzle had been removed at station b. Second, the supersonic diffusers of the inlets were at least 50 percent shorter than that of the diffuser apparatus. This factor probably was favorable for the recoveries of the diffuser apparatus. Significant increases of pressure recovery were obtained in reference 18 when a portion of the side-wall boundary layer was removed near the inlet throat. Similar increases in recovery could probably be obtained from the diffuser apparatus. For wind-tunnel applications, removal of this air would require more pumping facilities and might not be practical.

Flow characteristics in the supersonic diffuser.- The static-pressure distributions measured on the top wall of the supersonic diffuser at maximum recovery conditions and the corresponding shock-wave patterns are shown in figures 16 and 17. The pressures which were measured with no model in the airstream (fig. 16) are about the same as those computed by the oblique shock theory. These pressures are indicative of the maximum static-pressure rise in the supersonic diffuser since they were obtained at minimum values of A_c/A_b. The pressure rise (ratio of final to initial static pressure) varies from 3.6 at $M_\infty = 2.50$ to 12 at $M_\infty = 4.50$. It seems probable at $M_\infty = 4.50$ that the orifice at station 51.3 was too far upstream to measure the entire rise due to the last shock reflection and that the actual pressure ratio may have been about 20. These measured rises are significantly greater than the rise which could have been tolerated without separation at one shock reflection (fig. 5), and indicate the importance of the long supersonic diffuser for reducing the Mach number at the normal shock.

The computed shock-reflection points are indicated by the calculated pressure distributions. The photographs (fig. 16) indicate that the

actual reflection points are upstream of the predicted reflections as
might be expected since the boundary-layer growth tends to increase the
wall angle and oblique shock angle, and decrease the Mach number at any
given point. Some indication of the thickness of the boundary layer may
be gained by observing the shock pattern near the wall reflection points.

The distributions and shadowgraphs of figure 16 do not show conclu-
sively the factors which govern the minimum running-contraction ratios
of the supersonic diffuser. It was observed throughout the entire Mach
number range that with no model the minimum values of contraction ratio
were reached when the fourth oblique shock reflection moved upstream
past station c into the supersonic diffuser. The significance of this
observation, if any, is not known at present. It was not observed to
occur when the model was in the airstream (fig. 17).

Although conclusive proof is not available it would seem probable
that the minimum contraction ratios are probably determined by some
choking phenomena at station c. This choking may be caused by boundary-
layer separation or by too much geometric contraction of the supersonic
diffuser. The latter phenomena would be more likely to occur at low test-
section Mach numbers. Measurement of the shock-wave angles at the end
of the supersonic diffuser indicates, however, that at low test-section
Mach numbers the flow outside the boundary layer is still appreciably
above a Mach number of 1 and should not choke at station c because of too
much geometric contraction. At the lower test-section Mach numbers it
is thought that boundary-layer separation on the moving walls would not
be likely to occur since the local and overall static pressure rises are
less relative to the pressure-rise data for separation (fig. 5) than at
the higher test-section Mach numbers. At the higher Mach numbers it is
more likely that the controlling factor could be separation on the moving
walls. It does not appear logical, however, that the separation can be
attributed directly to the fourth reflection moving into the supersonic
diffuser since this occurred throughout the Mach number range. Perhaps
the controlling factor throughout the Mach number range is the separa-
tion of the relatively thick side-wall boundary layer. Further studies
would be required to establish this fact.

With the model in the airstream the maximum static pressure rise at
each Mach number (fig. 17) was somewhat lower than without the model
because the minimum contraction ratios were larger. No calculated pres-
sure distributions are presented because of the complex shock pattern in
the diffuser. The minimum contraction ratios now occur before the fourth
shock reflection moves upstream of station c.

Some details of the side-wall boundary-layer flow may be deduced from
figure 17(a) from the oil-flow pattern on the window. The effect of the
static-pressure distribution in deflecting the wall boundary layer toward

the diffuser center line is evident in the·upstream portion of the pic-
ture. Further downstream a small area of separation apparently exists
near the wall center line where the shock waves from the top and bottom
wall intersect. The existence of this local area of separation at a
relatively low static-pressure rise is probably a result of the thick
boundary layer which exists on the side-wall center line because of the
reduction of side-wall area and the static-pressure effects previously
noted. The piling up of the boundary layer on the side-wall center line
has been noted in reference 18.

Starting characteristics.- The starting contraction- and compression-
ratio characteristics of the basic diffuser apparatus are shown in fig-
ures 18 and 19. These data were obtained with atmospheric back pressure.

The starting compression ratios are a function both of the model in
the airstream and the contraction ratio of the supersonic diffuser. The
latter effect has been previously noted (ref. 20, for example). With no
model in the airstream the minimum starting-compression ratios at $M_\infty = 4.0$
and above were less than the theoretical values which were based on a
normal-shock recovery at test-section Mach number. As mentioned in ref-
erence 20 the separation in the supersonic nozzle apparently reduces the
test-section Mach number during the starting process and thereby reduces
the starting-compression ratio.

The starting-contraction ratios (fig. 19) are independent of Reynolds
number, but are affected by the model in the airstream. These ratios are
less than the theoretical values because of the reduced test-section Mach
number which occurs during the starting process. This effect was also
noted in reference 20.

With Injection

Pressure recovery.- The basic data which were obtained with the
injector at station 61.8 are presented in figures 20 and 21. In the top
portions of figure 20 the data obtained with no model in the airstream
are plotted as a function of the contraction ratio A_f/A_e of the second
supersonic diffuser for various values of A_c/A_b. These data have been
cross-plotted on the bottom of figure 20 as a function of A_c/A_b. The
computed pressure recoveries were obtained by the use of the injector
equations presented in reference 7. For these computations, Mach num-
ber and total pressure at station c were determined by oblique shock
theory, and the normal shock was assumed to occur at station f. In fig-
ure 21 the pressure-recovery data which were obtained with the model in
the airstream are presented.

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The pressure recoveries obtained with no model in the airstream (fig. 20) vary with contraction ratio in a manner which is generally similar to that of the computed recoveries. The actual recoveries are about 0.05 to 0.17 lower than the computed results. The minimum running contraction ratios of the main supersonic diffuser A_c/A_b (fig. 14) were not significantly affected by the injector. Only a limited amount of data were obtained with the model in the airstream. These data, however, were obtained at near minimum running-contraction ratios and indicate the maximum recoveries to be expected under these conditions.

The injector data are summarized in figure 22. Maximum recoveries which were obtained with no model in the airstream and with minimum values of A_f/A_e varied from 0.64 at $M_\infty = 2.95$ to 0.56 at $M_\infty = 4.70$. These recoveries were reduced about 0.08 when the contraction of the second supersonic diffuser was eliminated $(A_f/A_e = 1.0)$, or when the model was placed in the airstream. With the model in the airstream and with $A_f/A_e = 1.0$, the maximum recoveries varied from 0.55 at $M_\infty = 2.95$ to 0.45 at $M_\infty = 4.45$.

The recoveries of the injector configurations are appreciably higher than the recoveries of the basic diffuser apparatus at the higher Mach numbers (fig. 22). At $M_\infty = 4.7$ the maximum recoveries with and without injector (with no model) were 0.56 and 0.21, respectively. The large increase in recovery may result from several factors. First, the total pressure of the injector air may be greater than the total pressure of the main airstream because of the viscous and oblique shock losses which occur in the main airstream. Hence, when the relatively large quantity of injector air (fig. 8) mixes with the main airstream, the total pressure of the merged streams may be greater than that of the main airstream with no injector. No surveys have been made to date to determine the total pressure of the main airstream at the injection station and no estimates of this mixing effect can therefore be made. Second, some aerodynamic compression of the main airstream may occur in the mixing section if the static pressure of injector air is greater than the static pressure of the main airstream. This effect, however, is thought to be small at maximum recovery conditions on the basis of the static-pressure measurements of figure 16. Third, the injector air may be acting as a boundary-layer control device by reenergizing in the mixing section the boundary layer of the top and bottom diffuser walls, and reducing the normal-shock boundary-layer interaction losses. It should be mentioned that with the injector configuration, the wall boundary layer at the normal shock is generally relatively thin since most of the wall surface originates at the injector. At the higher Mach numbers a large portion of the air enters the tunnel through the injectors and one might expect the recovery to be primarily dependent on the characteristics of the injected air. Hence, the measured recovery might be expected to be somewhat less than 0.63, the theoretical normal-shock recovery at the

injector Mach number of 2.19 and to be relatively independent of test-
section Mach number. With no model in the airstream and $A_f/A_e = 1.0$
the measured recoveries (fig. 22) varied from 0.53 to 0.48 as the test-
section Mach number varied from 3.9 to 4.7. As A_f/A_e becomes less
than 1.0 both the theoretical and experimental recoveries would be
expected to increase.

At the lower test-section Mach numbers the recoveries could undoubt-
edly have been improved by the use of lower injector Mach numbers. As
will be mentioned later the injector Mach number could not have been sig-
nificantly reduced at the higher test-section Mach numbers.

Flow details.- The flow details at near maximum pressure-recovery
conditions are shown in figures 23(a) to 23(d). The wake which leaves
the top surface of the main supersonic diffuser at station d gives an
indication of the static pressure of the diffuser and injector airstreams
at station d. Inasmuch as the axis of the injector was inclined 10°
inward with respect to the tunnel center line, equal pressures in the two
streams would be indicated (to a first approximation) by a wake inclined
at 5° toward the center line. At $M_\infty = 2.95$ the wake is inclined away
from the tunnel axis indicating that the diffuser air expands at station d
and the injector air is compressed by a significant amount. At $M_\infty = 3.90$
and 4.30 the wake is inclined by slightly less or more, respectively,
than 5°, indicating that at these Mach numbers the static pressures of
the two streams are about equal. At $M_\infty = 4.70$, the pressure of the dif-
fuser air appears to be less than that of the injector. The small curved
shock (figs. 23(b) to 23(d)) which exists near the tunnel center line
between station e and f is thought to indicate that the side-wall boundary
has been separated by the shock waves generated by the second supersonic
diffuser at station e.

It was previously mentioned that at the higher Mach numbers the
injector Mach number was at a near-minimum value. This statement is
based on the flow phenomena shown in figures 23(d) to 23(g). The photo-
graphs indicate that as the static-pressure difference between the injec-
tor and main airstreams increased (A_c/A_b increased for a given M_∞), a
normal shock formed in the main airstream (fig. 23(f)). In figure 23(g)
the static-pressure difference was large enough to force the normal shock
to move upstream of station d. Further increases of the pressure dif-
ference (A_c/A_b increased to values greater than 0.270) forced the nor-
mal shock upstream of the diffuser into the supersonic nozzle. A similar
effect would be expected if the contraction ratio of the main supersonic
diffuser was maintained at a near-minimum value and the static pressure
of the injector air was increased (injector Mach number decreased).

Effect of Diffuser Configuration on Wind-Tunnel Performance

Pressure recovery.- The pressure recoveries of supersonic wind tun-
nels in which the normal shock occurs at approximately test-section Mach
number (no supersonic diffuser) are about 70 to 85 percent of the normal-
shock recoveries (fig. 24). Data presented in references 7 and 8 indicate
that these basic recoveries may be significantly improved by the use of

air injection $\left(0.9 = \dfrac{w_j}{w_{ts}} = 2.2\right)$ just downstream of the test section to

reduce, mainly by aerodynamic compression, the normal-shock Mach number.
Still larger increases of the basic recoveries have been obtained in ref-
erences 2, 3, 4, and 6 by the use of a variable-geometry supersonic dif-
fuser to reduce the normal-shock Mach number.

The recoveries obtained with the injector at the end of the super-
sonic diffuser are much higher at the higher Mach numbers than any pre-
viously reported recoveries for wind-tunnel diffusers. Conversely, the
compression ratios are lower. At $M_\infty = 5.0$, for example, the compres-
sion ratios required by a conventional normal-shock diffuser, a design
with variable-geometry supersonic diffuser, and a design with both a
supersonic diffuser and injector are about 20, 10, and 1.9, respectively.
For an existing tunnel with a conventional, fixed-diffuser design, and
a compression ratio of about 2.5, the advantages of modifying the dif-
fuser design to incorporate a variable-geometry supersonic diffuser and
injector are obvious. Taking into account the recoveries which may be
expected with a model in the airstream the available compression ratio
of the existing drive system should be sufficient to increase the test-
section Mach number from about 2.5 to 5.0. This assumes that the tunnel
may be started at low test-section Mach numbers.

This method of starting and operating a tunnel was demonstrated with
the diffuser apparatus. Supersonic flow in the test section was estab-
lished at $M_\infty = 2.5$, and the stagnation pressure increased to 37 pounds
per square inch absolute to equal a compression ratio of 2.5 with atmos-
pheric back pressure. Then the Mach number was continuously increased
to 4.5 without increasing the stagnation pressure. As the Mach number
increased, the values of the contraction ratios A_c/A_b and A_f/A_e
were decreased according to a preset schedule. It was later determined
by measuring the pressure recoveries that the required compression ratio
for this demonstration had not exceeded 2.0, although the Reynolds num-
bers were relatively low. During this demonstration the model support
strut was in the airstream.

The recoveries of the subject diffuser with no injector were about
0.05 to 0.15 higher than the recoveries of other configurations with

variable-geometry supersonic diffusers. Analysis of the data of ref-
erence 6 (which had a configuration very similar to subject diffuser)
indicates that at equal contraction ratios the recoveries of the subject
diffuser and the diffuser of reference 6 were about equal. The differ-
ence in maximum recoveries is apparently primarily due to a difference
in the minimum running contraction ratios. This difference may be due
to the lower Reynolds numbers of the tests of reference 6. For this to
be true, however, the Reynolds number effects must become proportionately
larger as the numbers approach those of reference 6.

Power.- The power requirements of a wind tunnel are a function not
only of the pressure recovery but also of the weight flow of air in the
tunnel circuit. Since the high recoveries of the injector configuration
were obtained with large air-flow quantities this configuration may not
be an optimum as far as power consumption is concerned. This fact is
illustrated in figure 25 by the computed power characteristics of several
hypothetical wind tunnels having the pressure recovery and weight-flow
characteristics shown in the figure.

Three hypothetical tunnels are considered. One tunnel has the char-
acteristics which might be obtained from an existing relatively low Mach
number tunnel which has been modified to incorporate a variable-geometry
supersonic diffuser and injector (tunnel 1, fig. 25). The total mass
flow was assumed to be constant over the Mach number range (curve n,
fig. 25) and the recovery (curve a) equal to that obtained during the pres-
ent investigation with a model in the airstream. A second tunnel (tun-
nel 2) was assumed to have the same recovery as the first tunnel but to
have a compressor capable of matching the air requirements of the constant-
area test section and the injector of the present investigation
(curves o + p). The third tunnel (tunnel 3) incorporates a variable-
geometry supersonic diffuser and has a compressor system which can match
the air requirements of a constant-area test section (curve p). The
recoveries of this configuration were assumed equal to those of the pres-
ent investigation (curve b). The reference tunnel is a conventional
design with a recovery equal to 85 percent of the normal-shock recovery
and with compressors matched to supply only the air required by the test
section.

If the reference tunnel were modified to incorporate a variable-
geometry diffuser without an injector, the power consumption would be
approximately one-half because of the higher pressure recoveries. For
large facilities this would amount to a very significant reduction of
operating costs. Further modification of the reference tunnel and drive
system to include the injector studied in the present investigation
results in less power savings than the initial modification. This con-
clusion might be altered if significant increases in recovery could be
obtained with low injector mass flows. As previously mentioned, the use
of the variable-geometry diffuser and injector permit conventional

tunnels designed for a Mach number of 2 to operate at Mach numbers up
to about 5. The relative power consumption, however, becomes very
large at the higher Mach numbers because of the constant airflow through-
out the Mach number range.

CONCLUSIONS

An investigation has been initiated to study methods for improving
the pressure-recovery characteristics of two-dimensional, variable-
geometry, supersonic-subsonic diffusers. The recovery characteristics
of the basic supersonic diffuser configuration and of a configuration
with an injector at the end of the supersonic diffuser have been deter-
mined at Mach numbers from 2.5 to 4.75, and at stagnation pressures from
40 to 205 pounds per square inch absolute. The test section was 6.75 inches
square. The Reynolds numbers based on the height of the test section and
a stagnation pressure of 120 pounds per square inch absolute varied from
12.2×10^6 to 4.0×10^6 for a Mach number range from 2.5 to 4.75. The
following conclusions have been obtained.

1. The pressure recovery of the basic configuration, which had a
relatively long supersonic diffuser to minimize shock boundary-layer
interaction effects, varied from 0.71 at a Mach number of 2.5 to 0.21 at
a Mach number of 4.75 for a stagnation pressure of 120 pounds per square
inch absolute. These recoveries are significantly higher than the cor-
responding normal-shock theoretical recoveries which vary from 0.50
to 0.075.

2. Analysis of the results from the basic configuration and of other
results from a similar configuration indicate that the effect of Reynolds
number on pressure recovery is negligible at a given contraction ratio of
the supersonic diffuser. As the Reynolds number decreases, the amount of
contraction which can be obtained in the supersonic diffuser decreases
and the pressure recovery therefore decreases. This effect was relatively
small for the Reynolds number range of the present investigation, but
appears to become larger at lower Reynolds numbers.

3. Use of a relatively high mass-flow injector (injector to main
stream mass flow ratio of 2.5 at a Mach number of 4.0) at the end of the
supersonic diffuser resulted in recoveries which varied from 0.65 at a
Mach number of 2.95 to 0.56 at a Mach number of 4.70.

4. The pressure recoveries of both configurations were reduced 0.05
to 0.10 by a typical research model and support system located at the end

of the supersonic nozzle. These recoveries were lower primarily because the minimum running-contraction ratios of the supersonic diffuser were larger.

5. The maximum test Mach number of existing supersonic wind tunnels with limited compression ratio and conventional normal-shock diffusers can be significantly increased by modifying the diffuser to incorporate a relatively long variable-geometry supersonic diffuser followed by an injector. For example, a compression ratio of 2.5 instead of about 20 is sufficient for a test-section Mach number of about 5.

6. The power consumption of high Mach number tunnels with conventional normal-shock diffusers can be significantly reduced by modifying the tunnel to incorporate a relatively long variable-geometry supersonic diffuser.

Langley Aeronautical Laboratory,
   National Advisory Committee for Aeronautics,
      Langley Field, Va., July 29, 1957.

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20. Moore, John A.: Investigation of the Effect of Short Fixed Diffusers on Starting Blowdown Jets in the Mach Number Range from 2.7 to 4.5. NACA TN 3545, 1956.
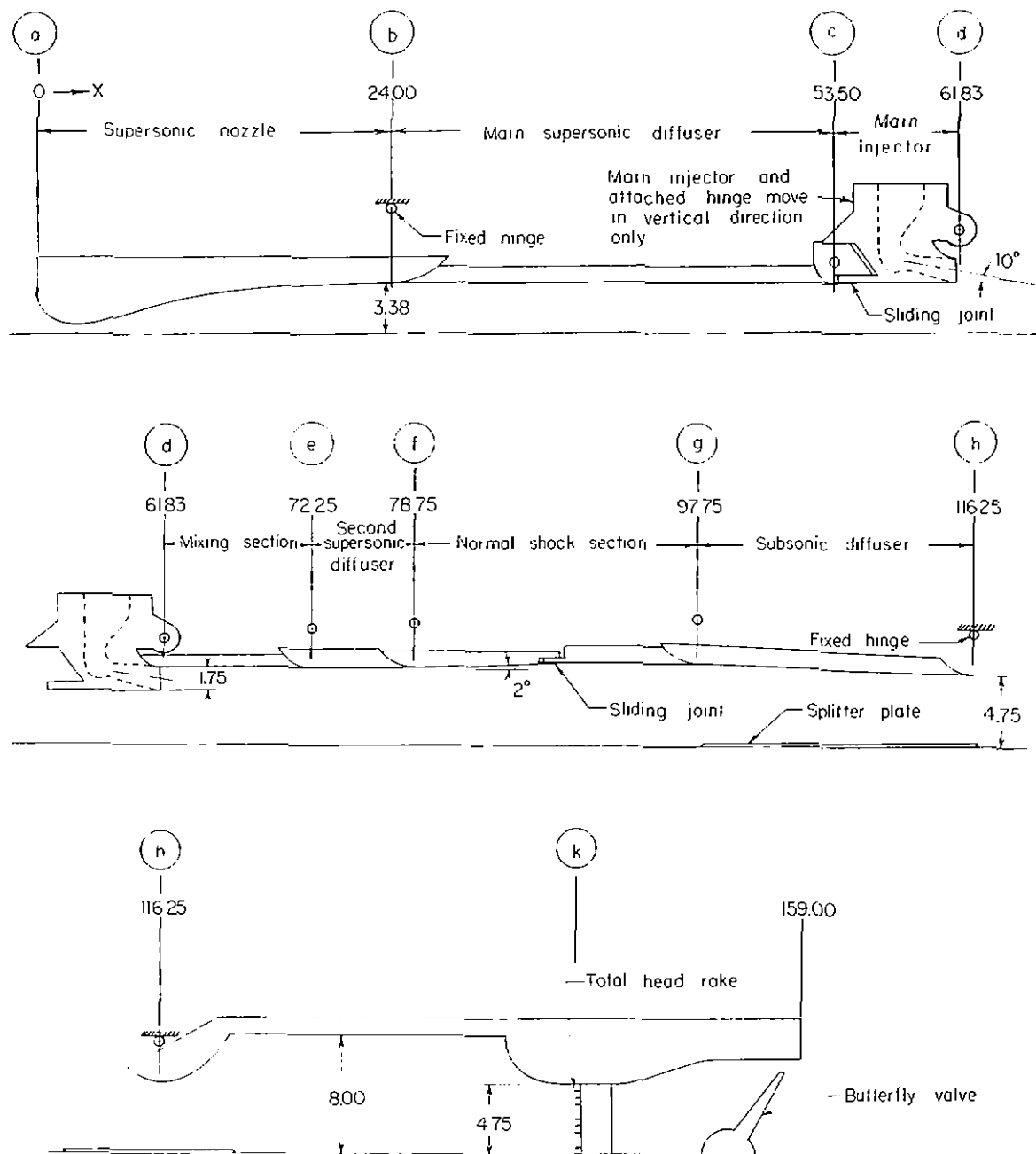
Figure 1.- Schematic drawing of variable-geometry diffuser apparatus.
All dimensions are in inches.
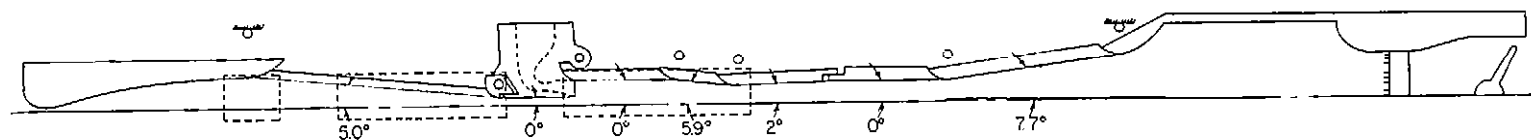
(a) $A_c/A_b = 0.240$; with main injector; $A_f/A_e = 0.738$.
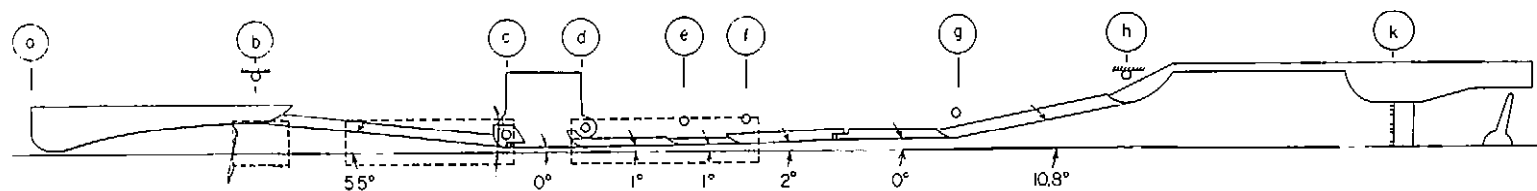
(b) $A_c/A_b = 0.166$; without main injector.

Figure 2.- Typical wall settings of variable-geometry diffuser apparatus.
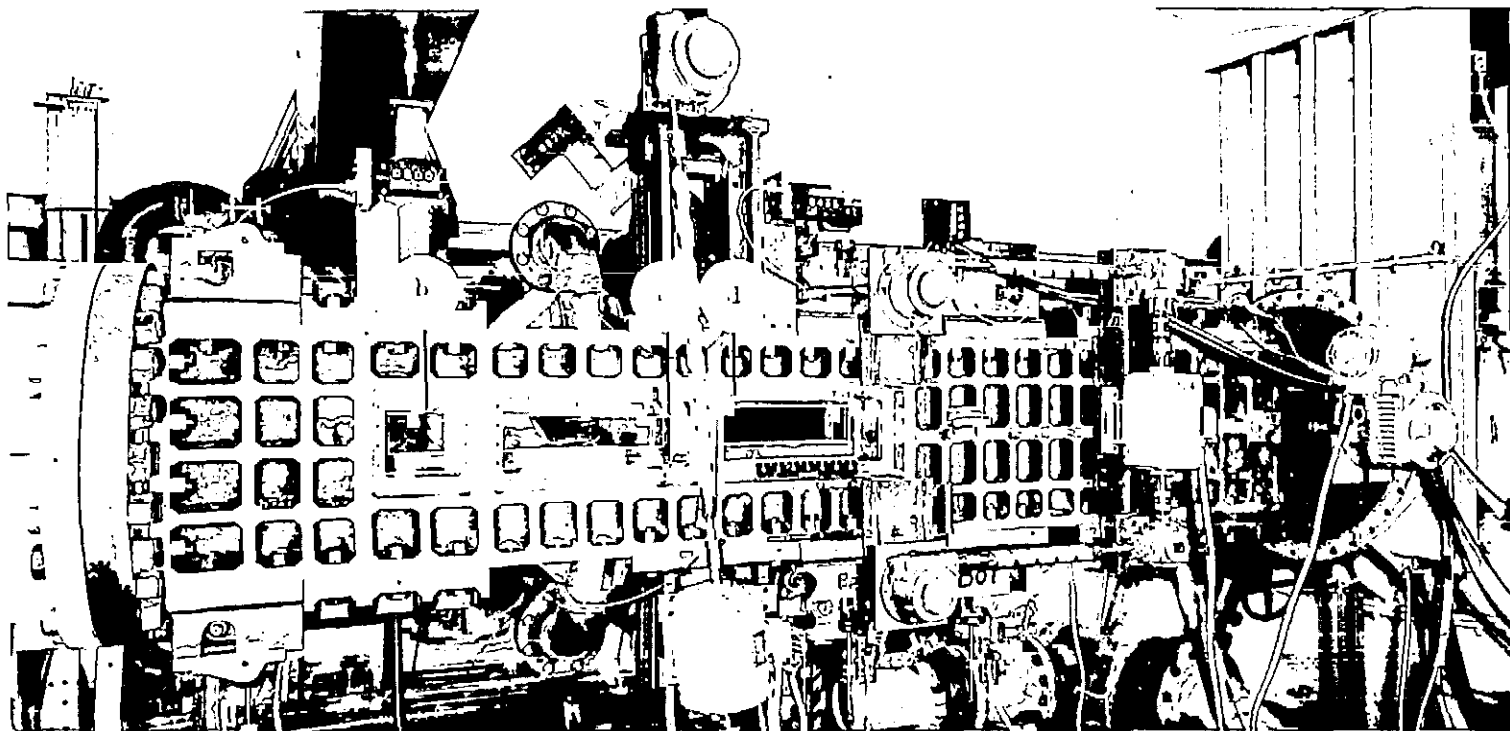
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Figure 3.- External view of variable-geometry diffuser apparatus.
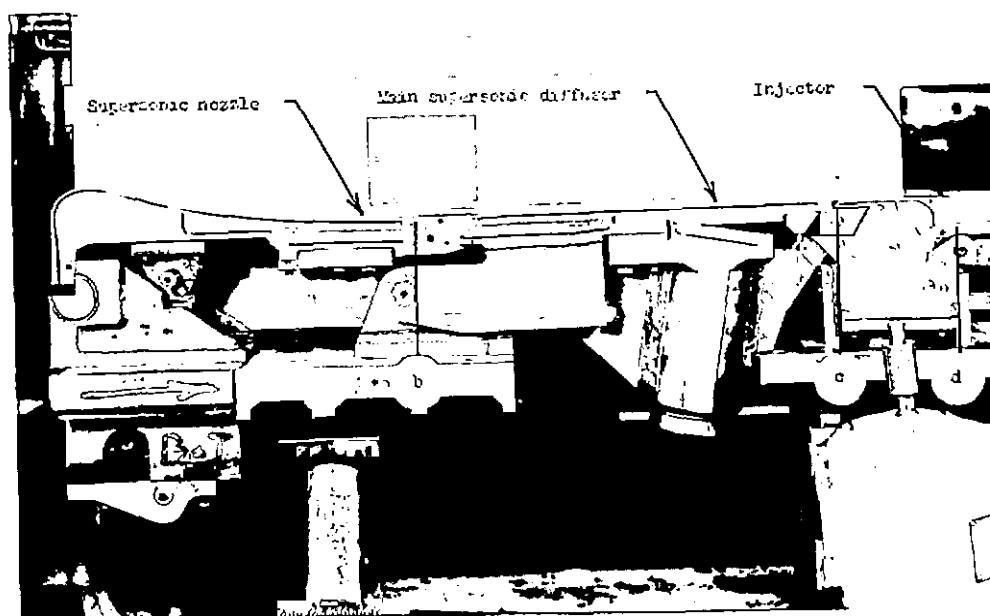
(a) Supersonic nozzle, main supersonic diffuser and injector sections of bottom wall.
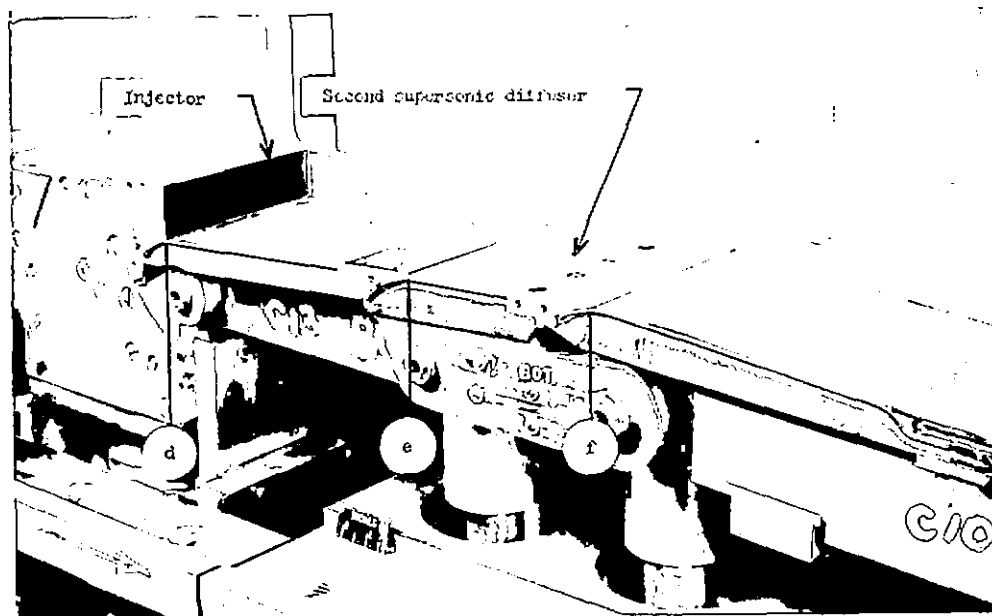
L-57-2727

(b) Injector, mixing, and second supersonic diffuser sections of bottom wall.

Figure 4.- Photographs of apparatus with one sidewall removed.
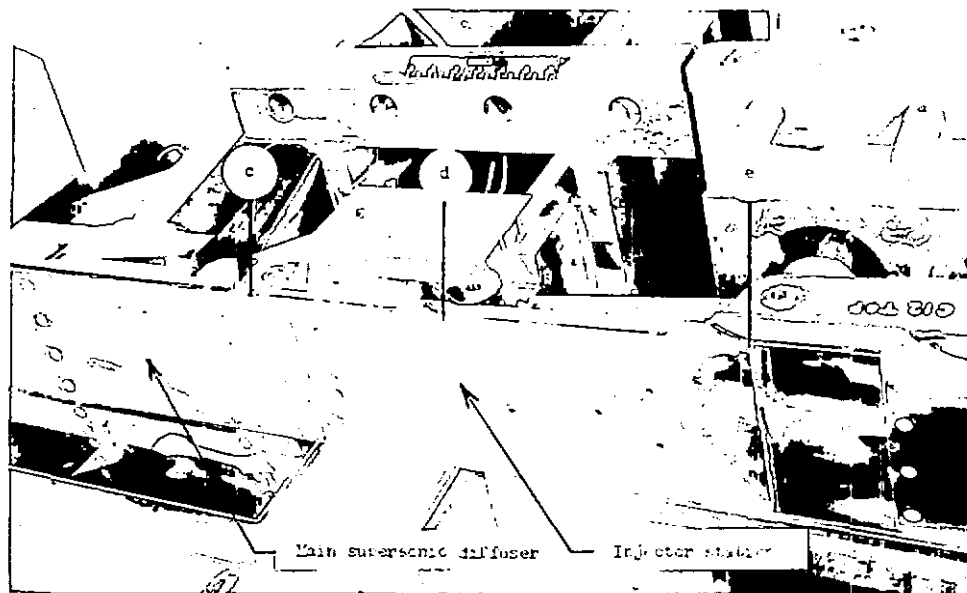
(c) Main supersonic diffuser and downstream sections of top
wall of configuration with no injector.
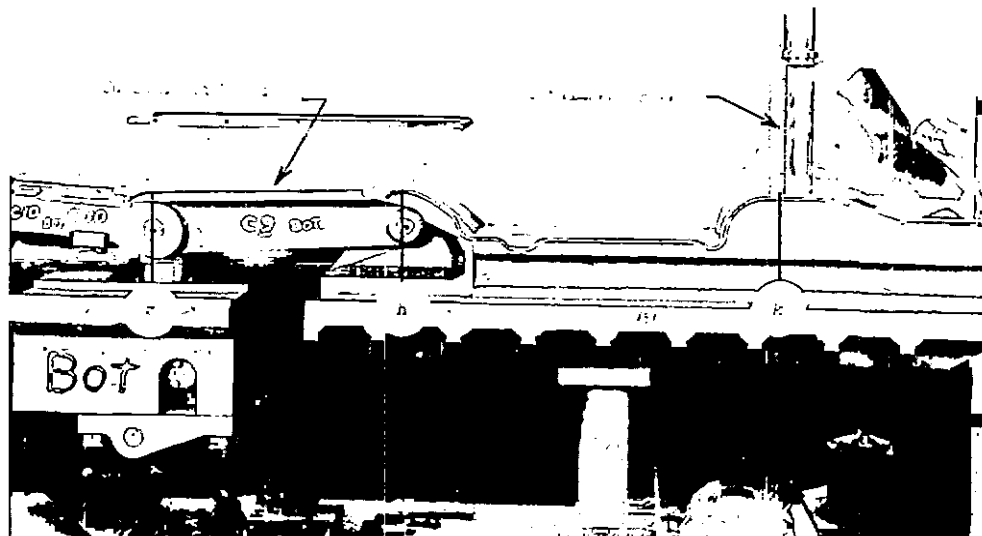
L-57-2728

(d) Subsonic diffuser and downstream end of apparatus.

Figure 4.- Concluded.
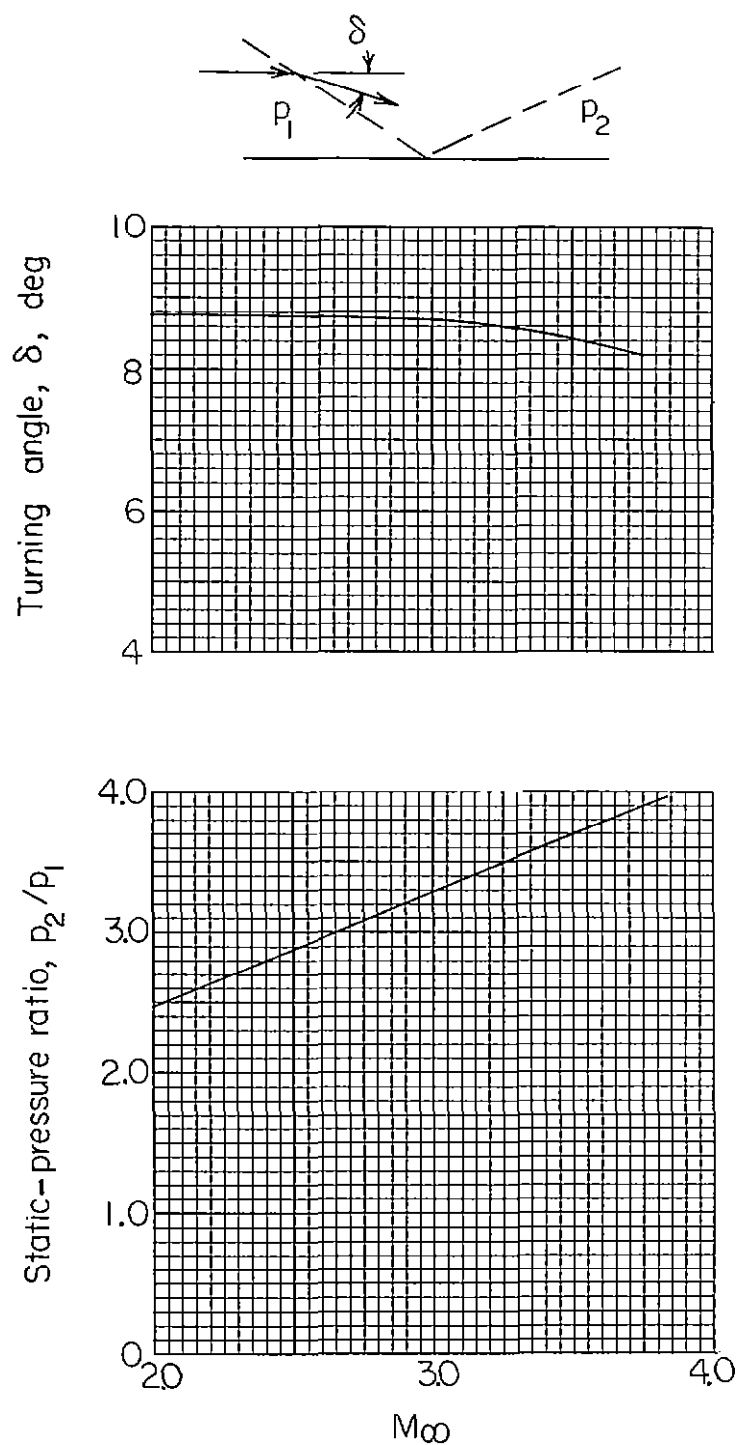
Figure 5.- Shock-wave characteristics which produce separation of a
turbulent boundary layer on a flat plate (ref. 12).
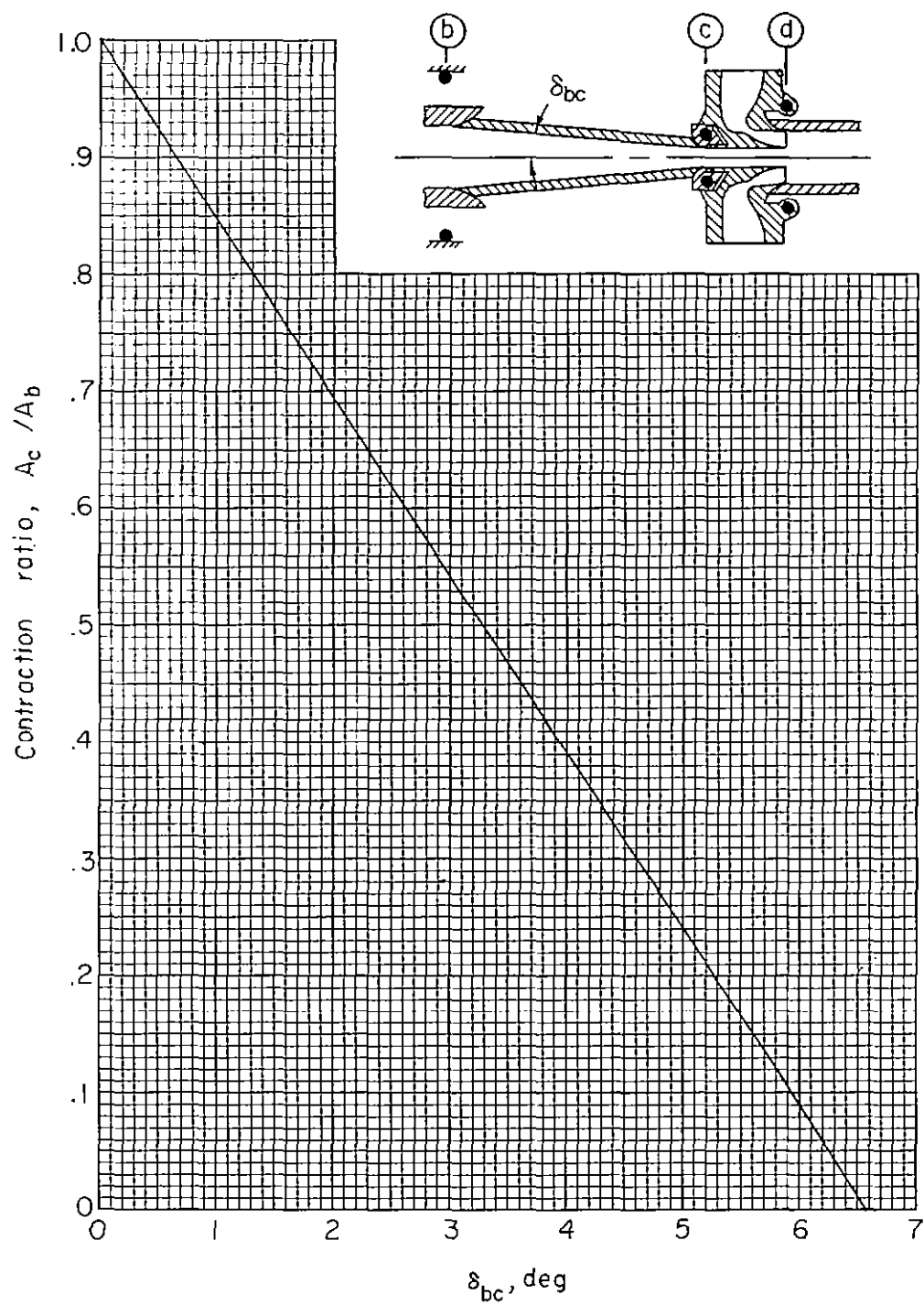
Figure 6.- Relationship between wall angle and contraction ratio of main supersonic diffuser.
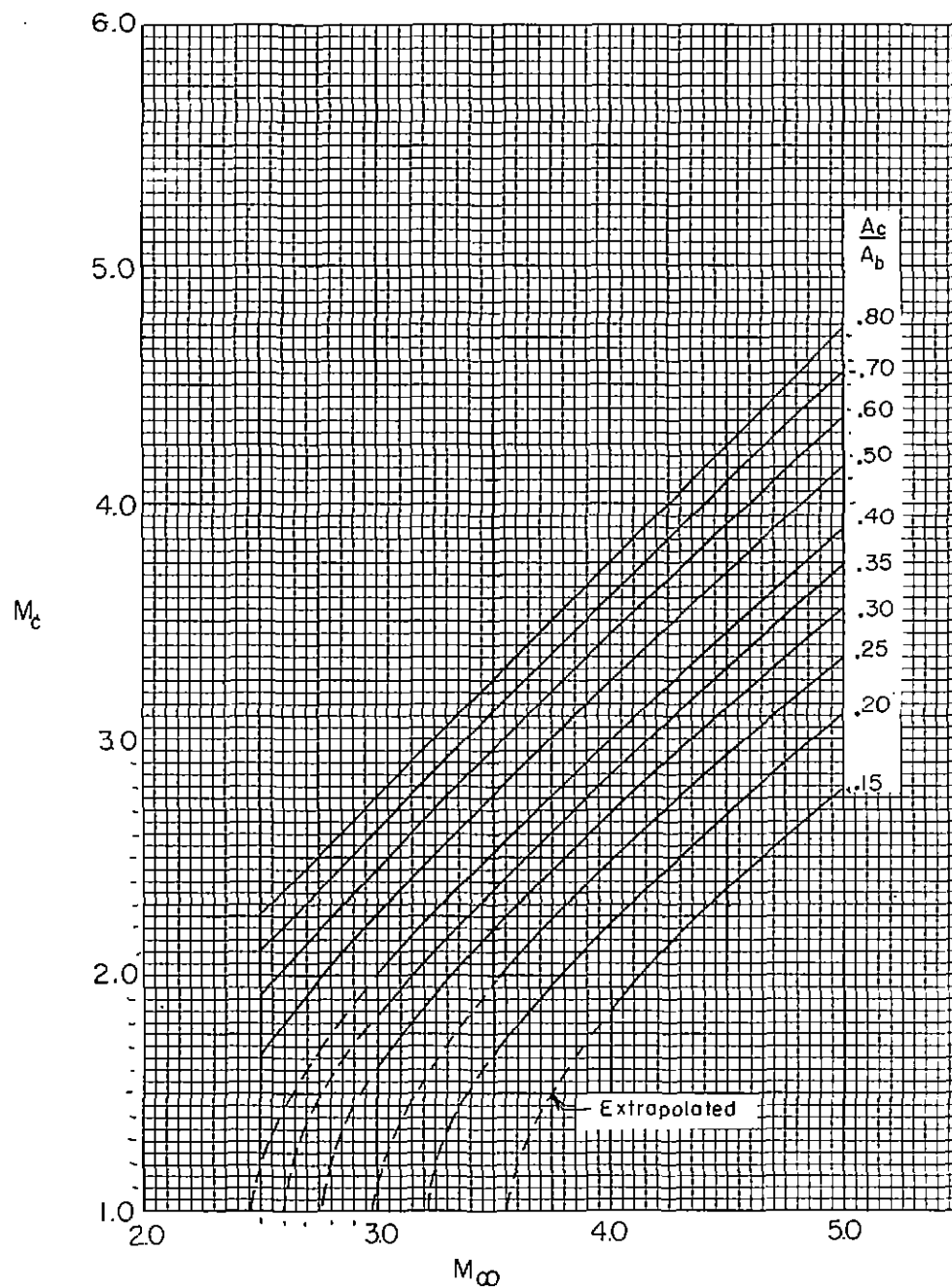
Figure 7.- Variation of computed Mach number at end of main supersonic diffuser with contraction ratio and test-section Mach number.
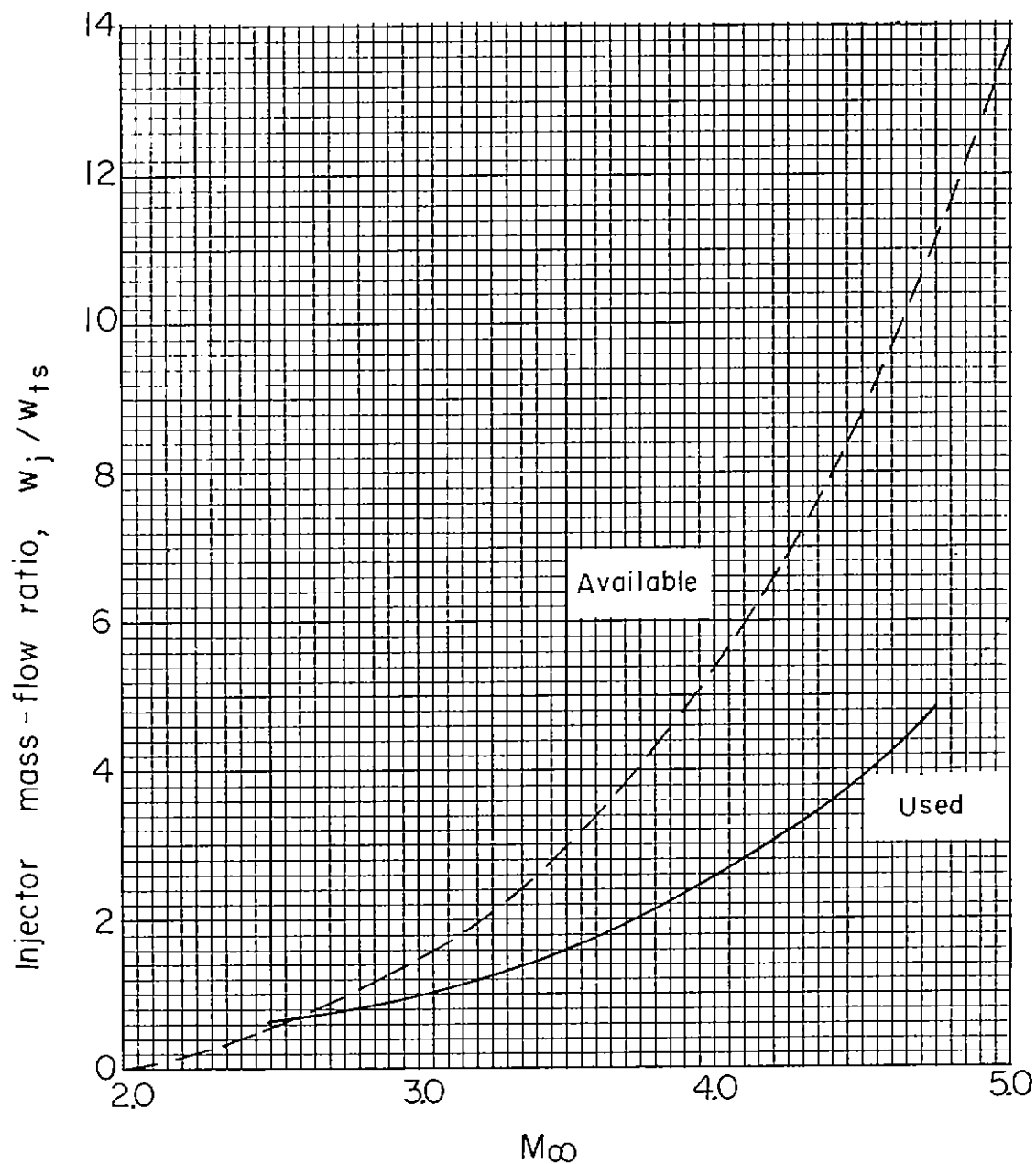
Figure 8.- Comparison of injector mass-flow ratio of diffuser apparatus
and mass-flow ratio available for a tunnel assumed to have constant
volume compressor and fixed-area test section above a Mach number
of 2.0.
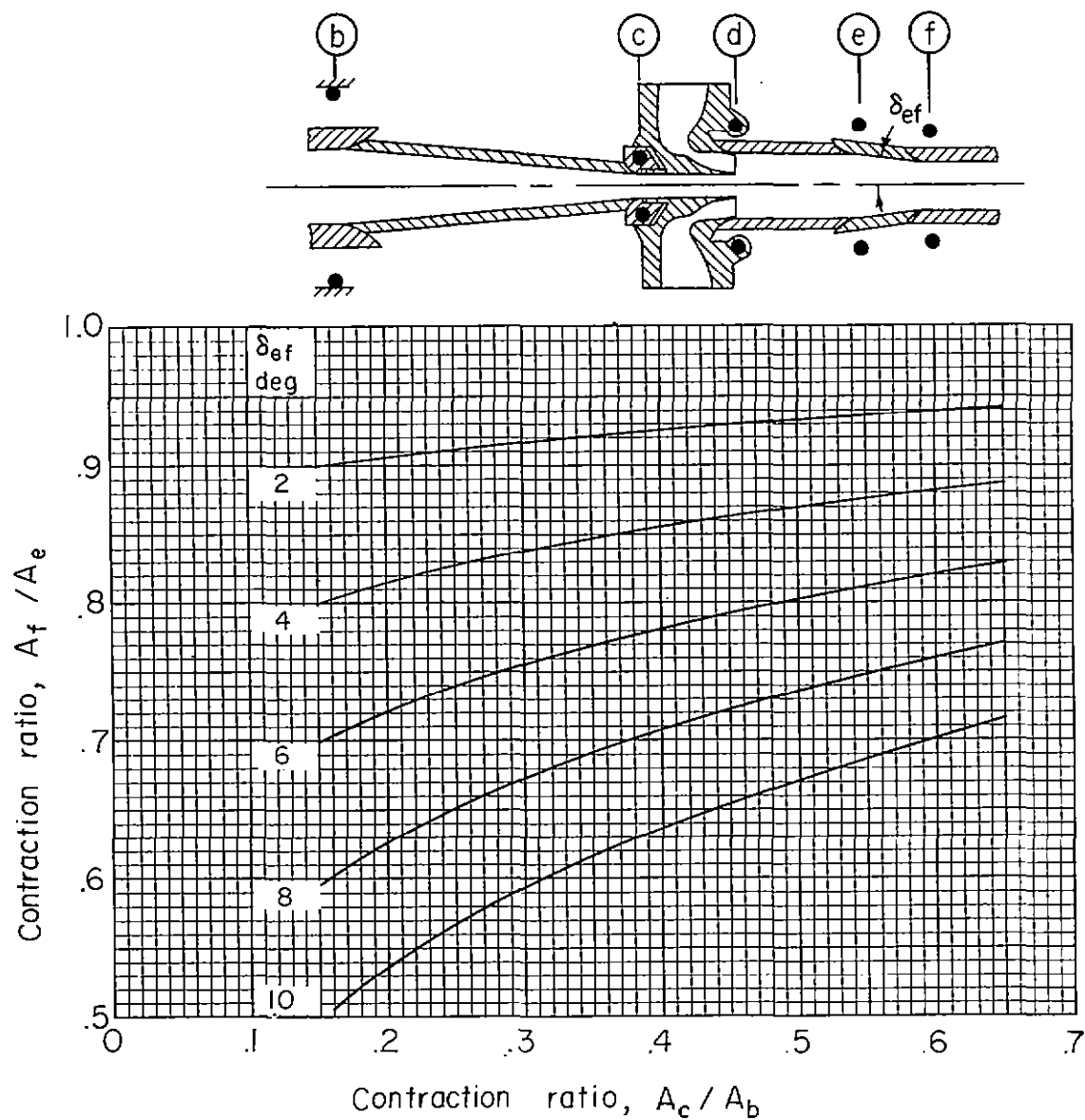
Figure 9.- Relationship between wall angle of second supersonic diffuser
and contraction ratios of main and second supersonic diffusers.

Station 24.0

18.25

15°

6.75

1.06

Sidewall

3.0

6.0

.375

1.25 DIA
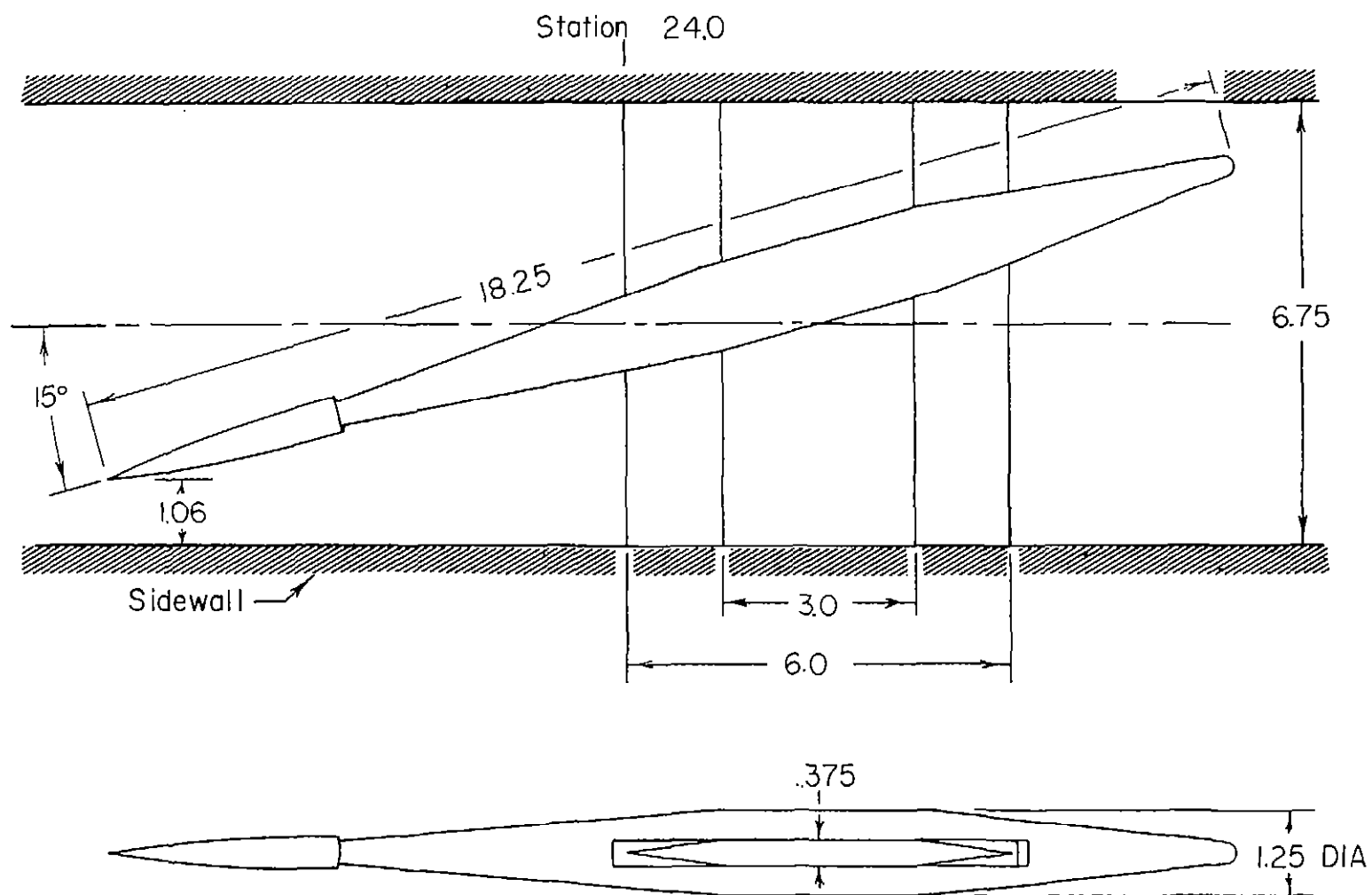
Figure 10.- Sketch of typical research model with its associated support system.   All dimensions
are in inches.
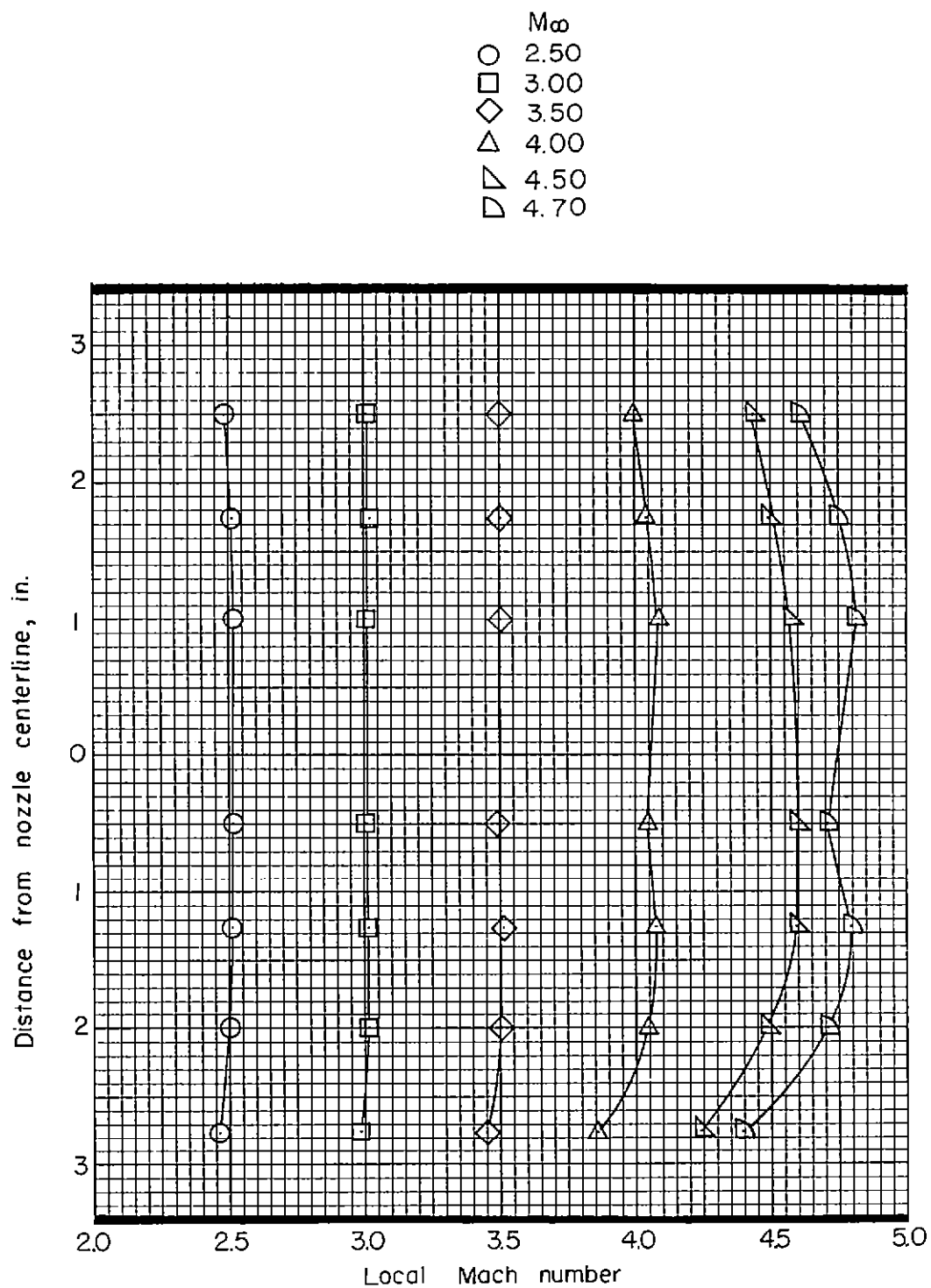
Figure 11.- Typical Mach number distributions at end of supersonic nozzle.
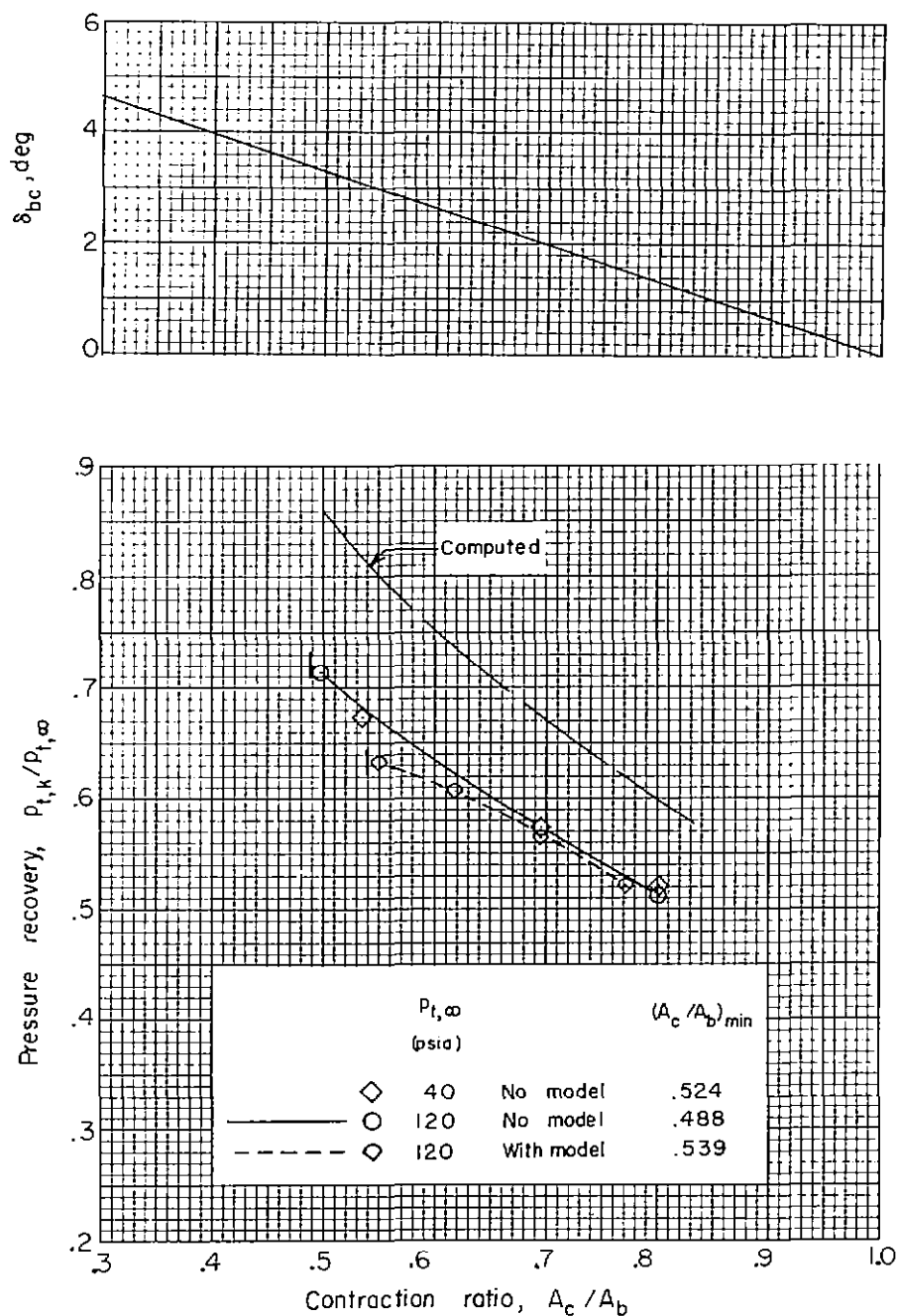
(a) M_∞ = 2.50.

Figure 12.- Effect of main-supersonic-diffuser contraction ratio on
pressure-recovery characteristics of diffuser apparatus without
injector.
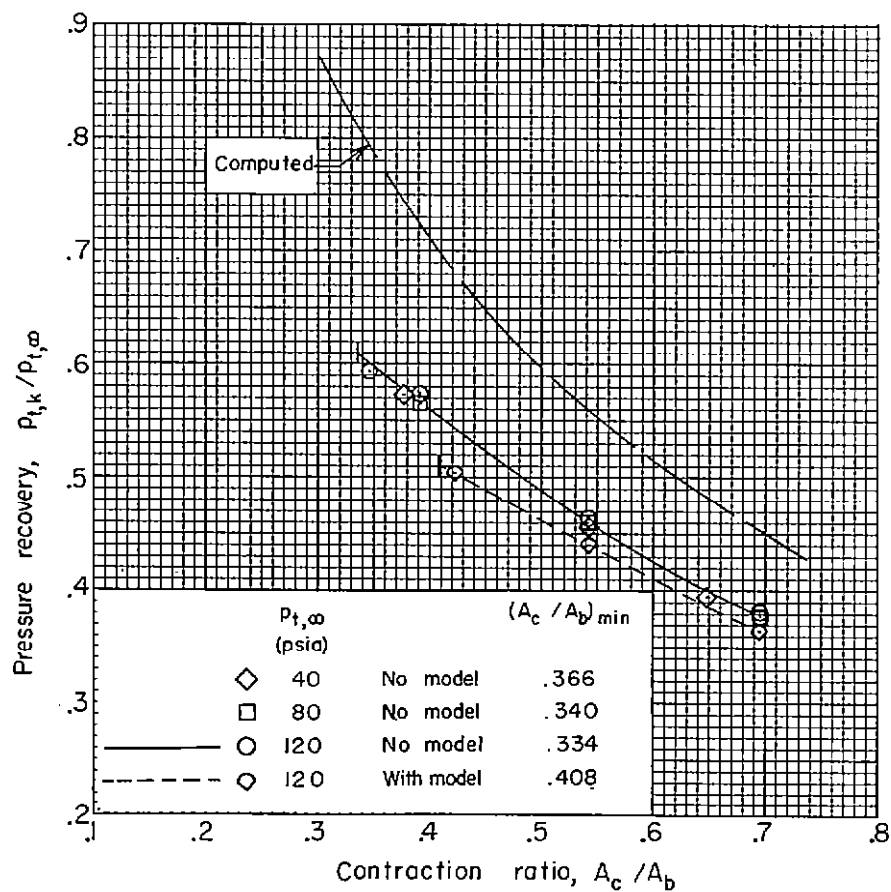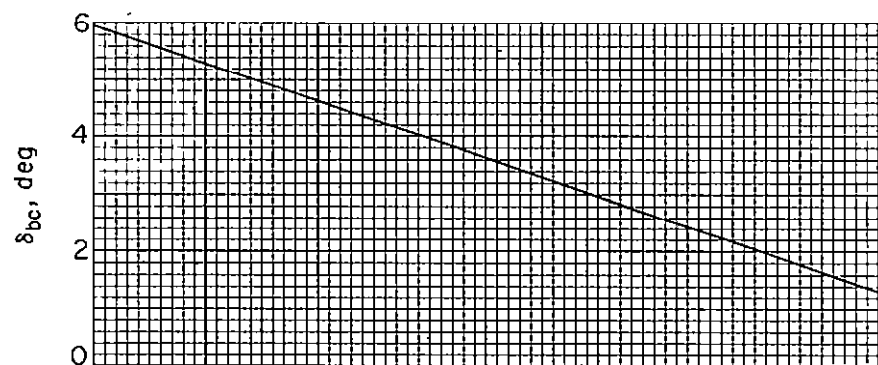
(b) $M_\infty = 3.00$.

Figure 12.- Continued.
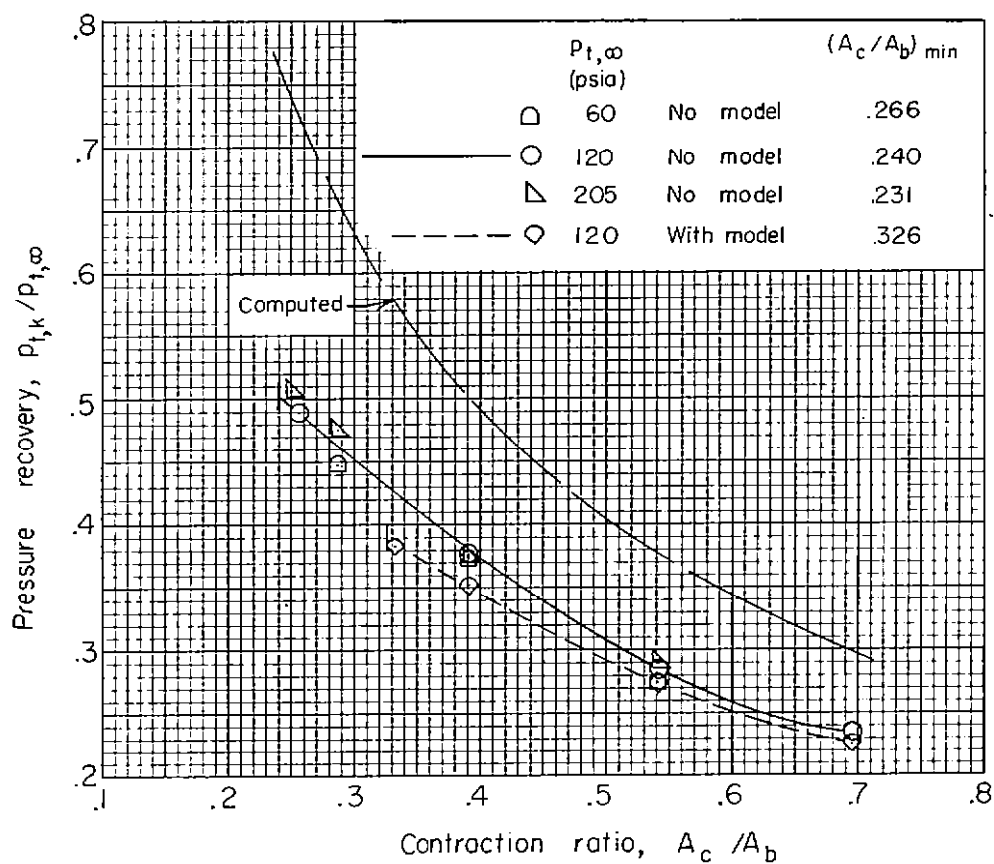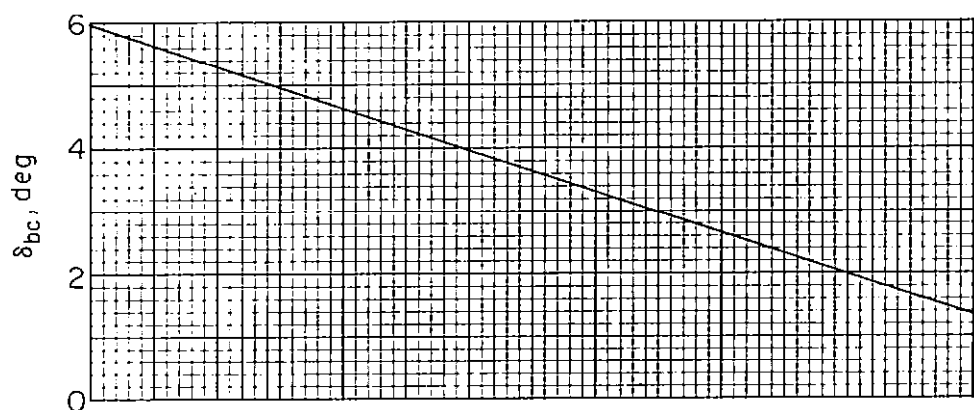
(c) $M_\infty = 3.50$.

Figure 12.- Continued.
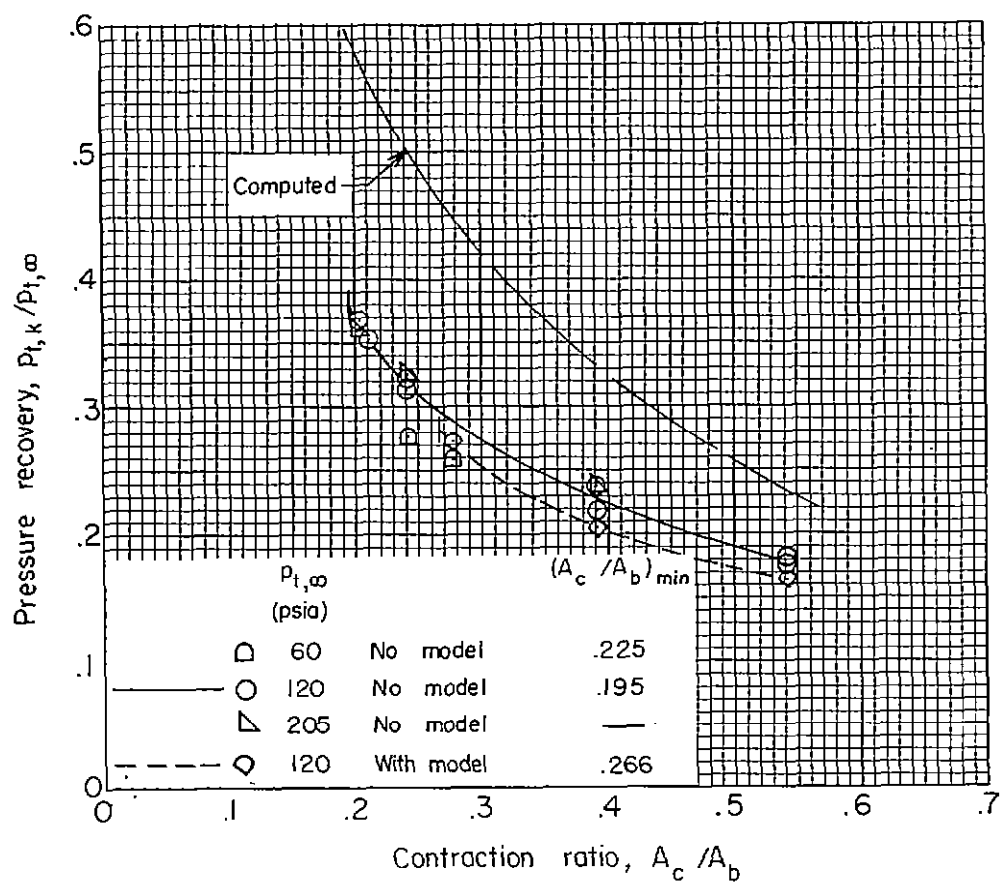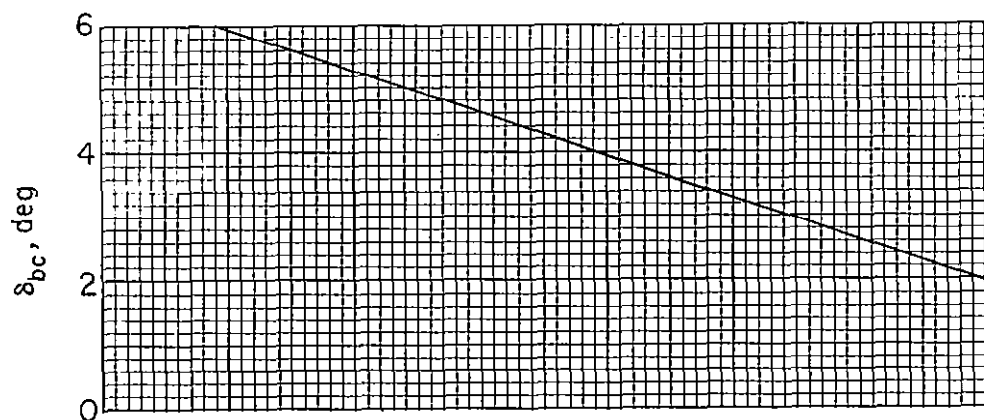
(d) $M_\infty = 4.00$.

Figure 12.- Continued.
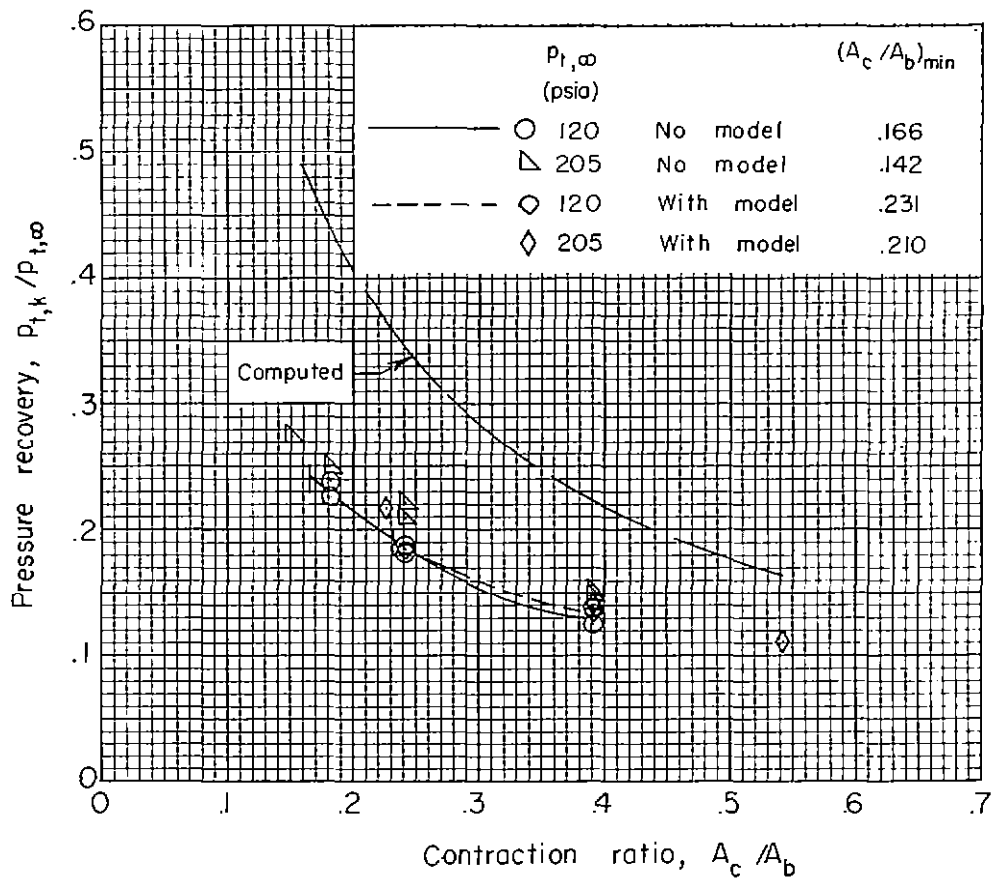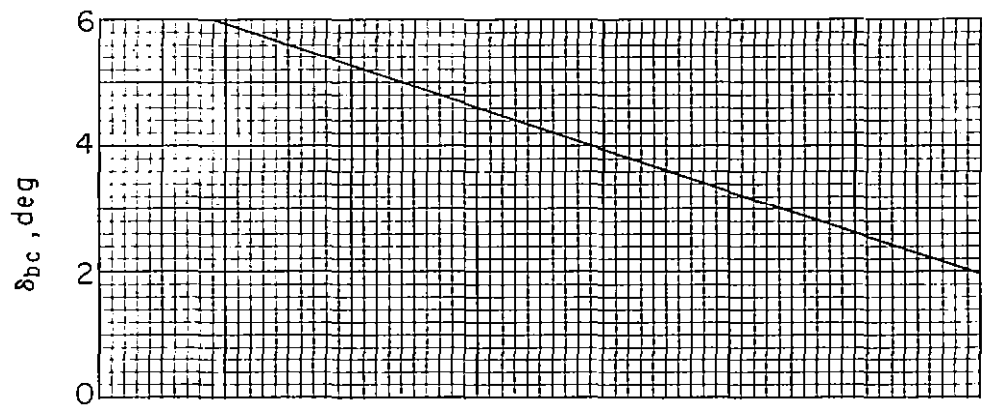
(e) $M_\infty = 4.50$.

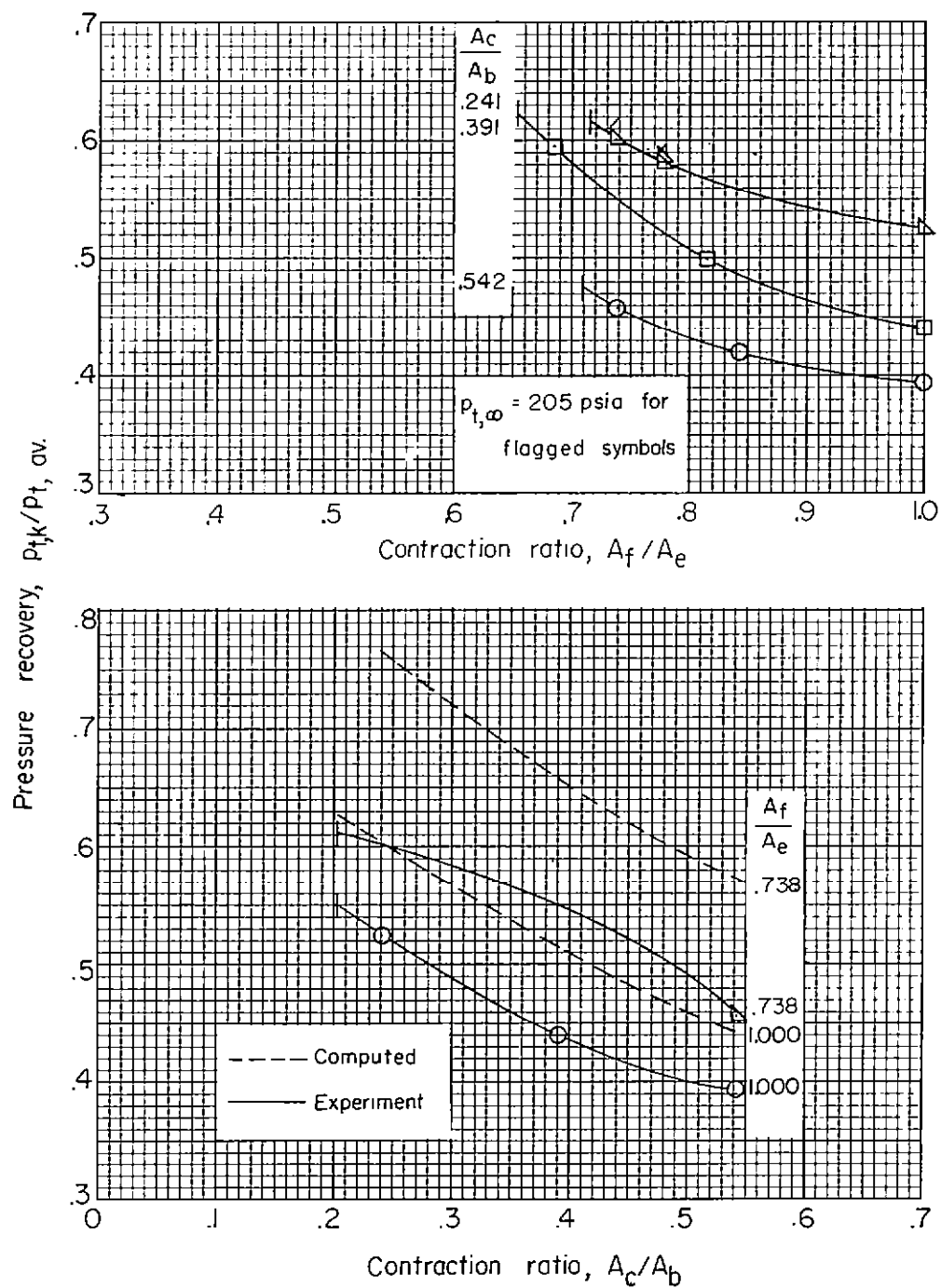Figure 12.- Continued.
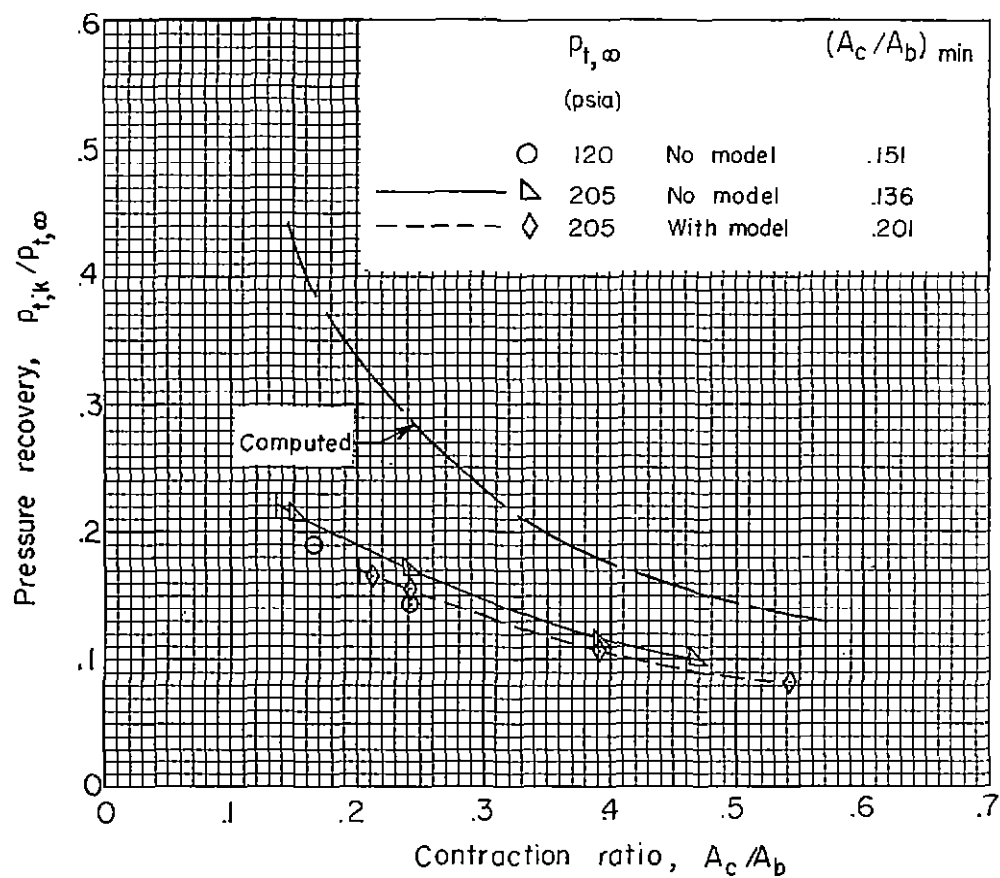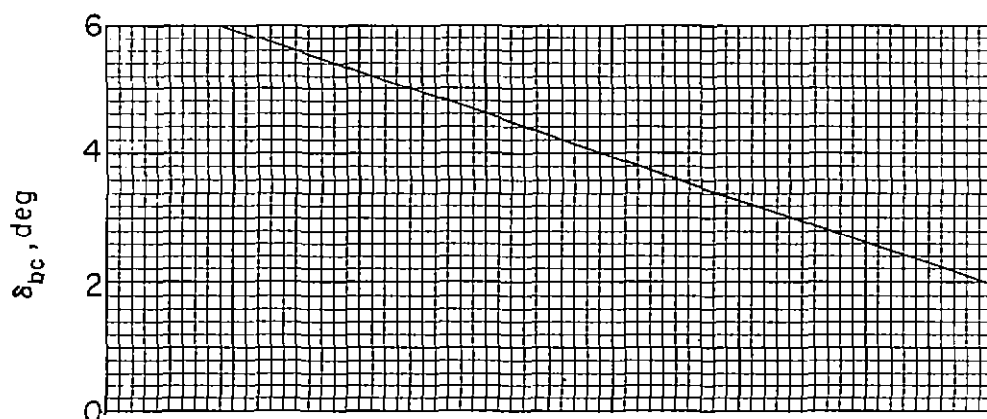
(f) $M_\infty = 4.75$.

Figure 12.- Concluded.
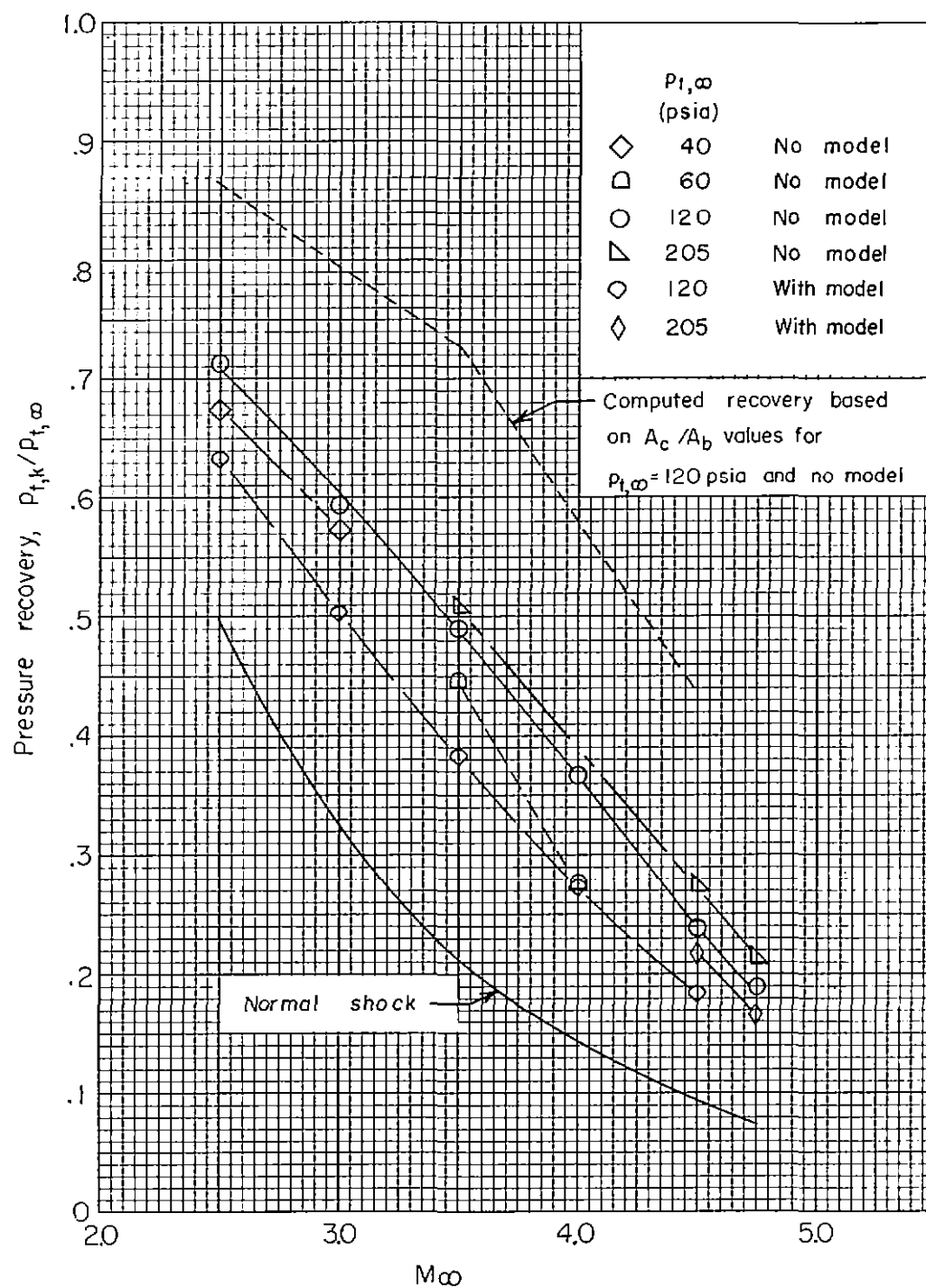
Figure 13.- Variation with Mach number of maximum measured pressure
recovery of diffuser apparatus without injector.
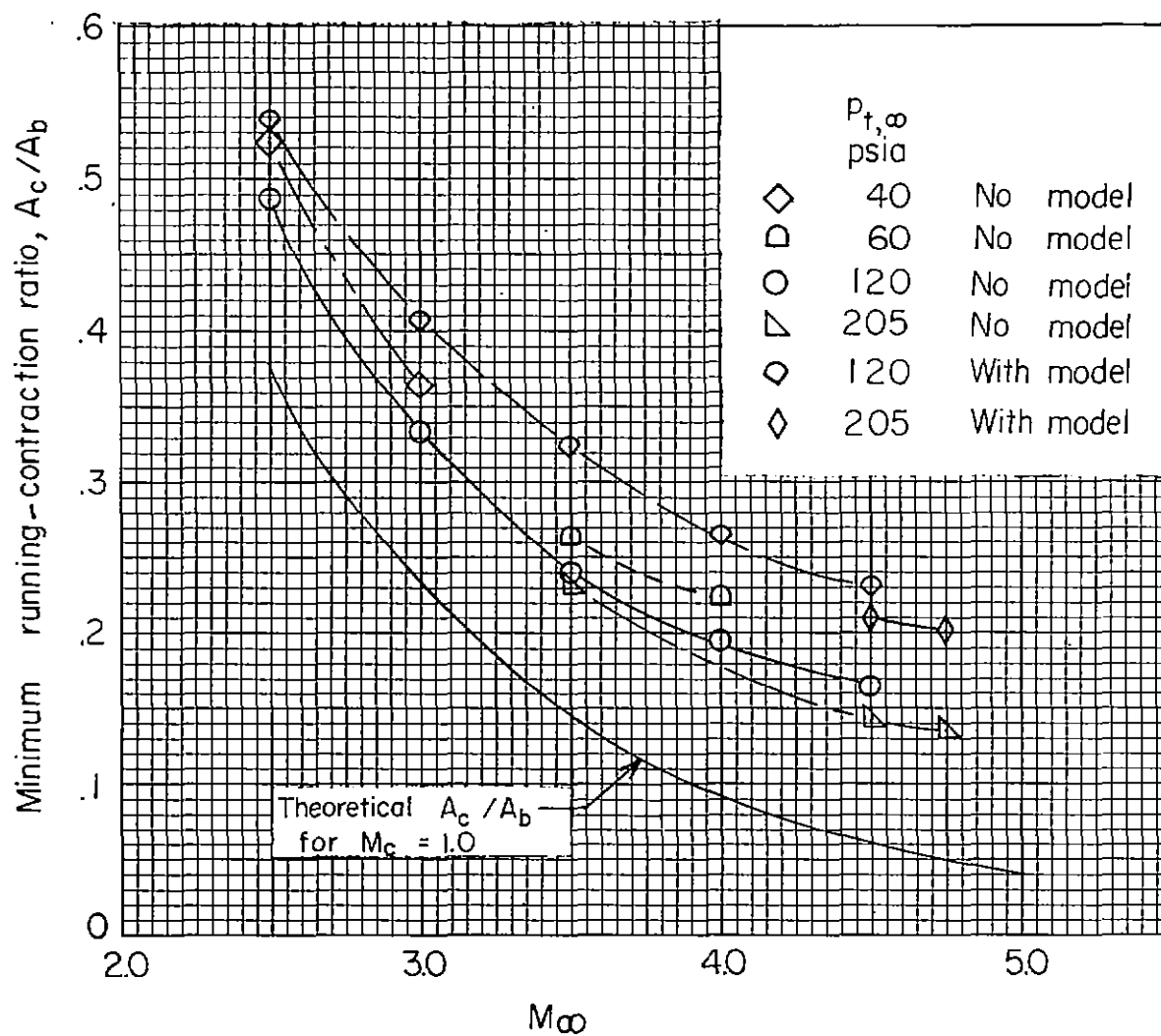
Figure 14.- Variation with Mach number of minimum running contraction ratio of main supersonic diffuser.
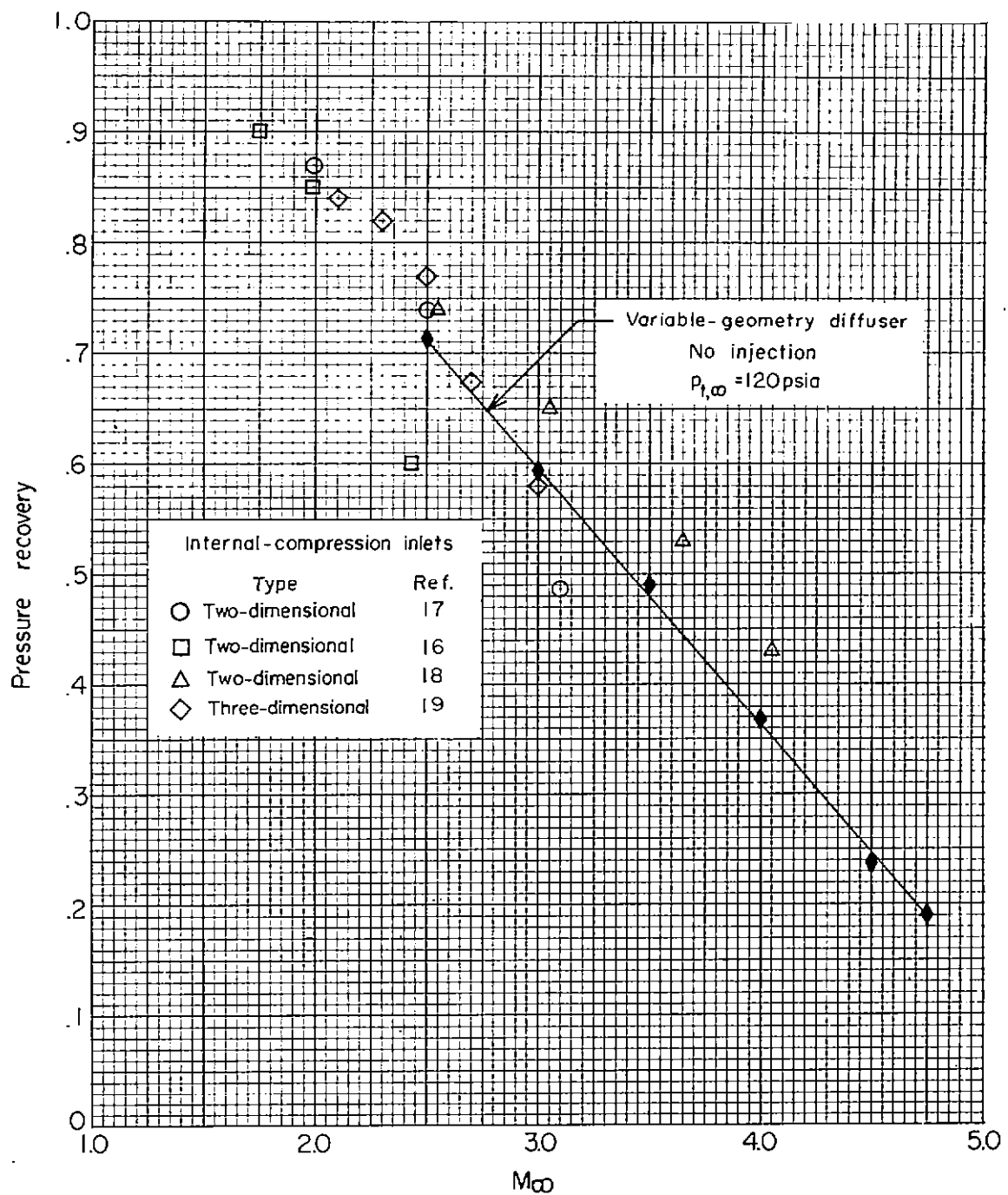
Figure 15.- Comparison of pressure-recovery characteristics of diffuser
apparatus and several internal-compression supersonic inlets with no
boundary-layer control.
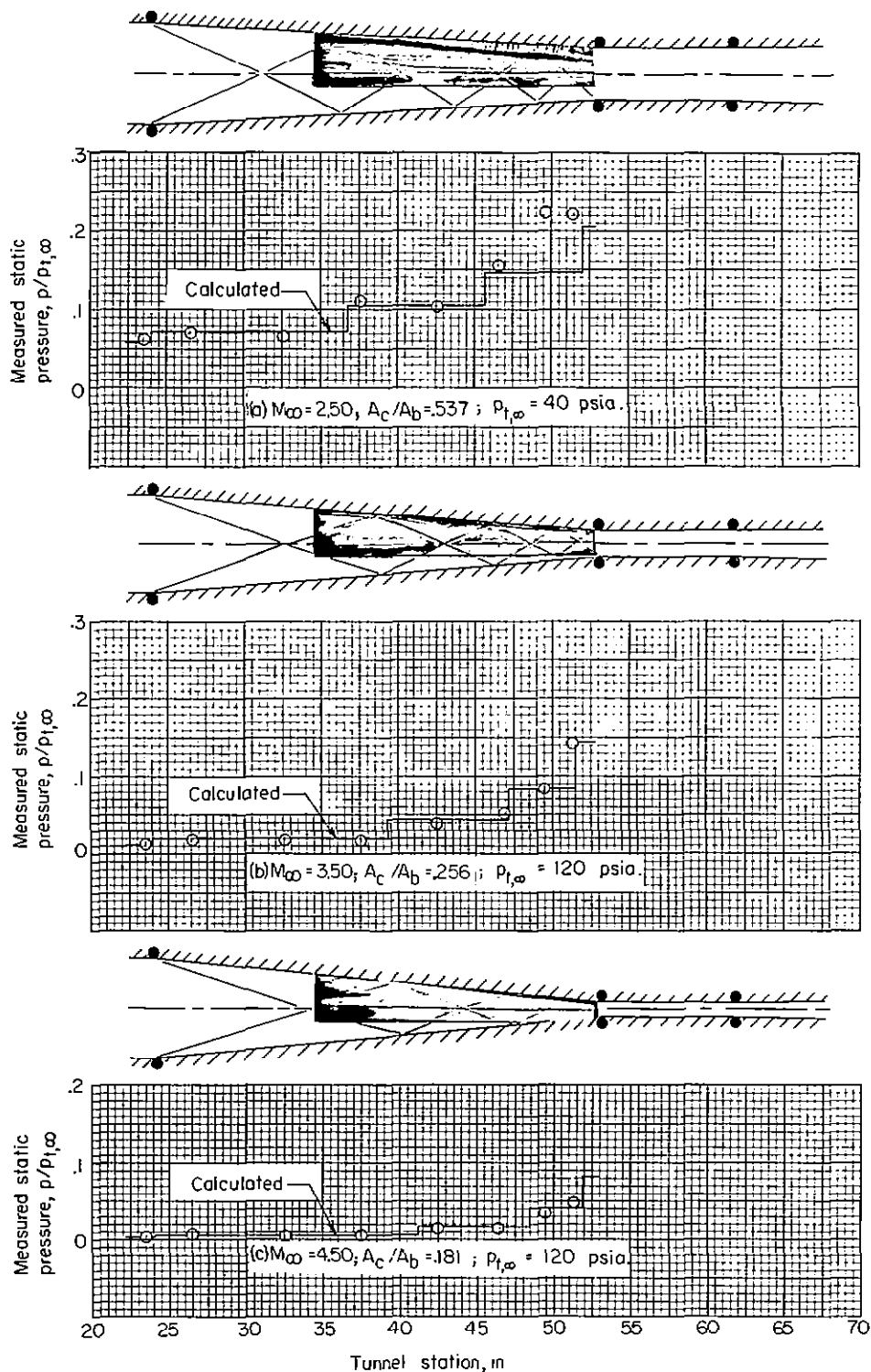
Figure 16.- Typical static pressure distributions on moveable wall and
flow patterns in main supersonic diffuser at near minimum values of
minimum-running contraction ratio. No model.
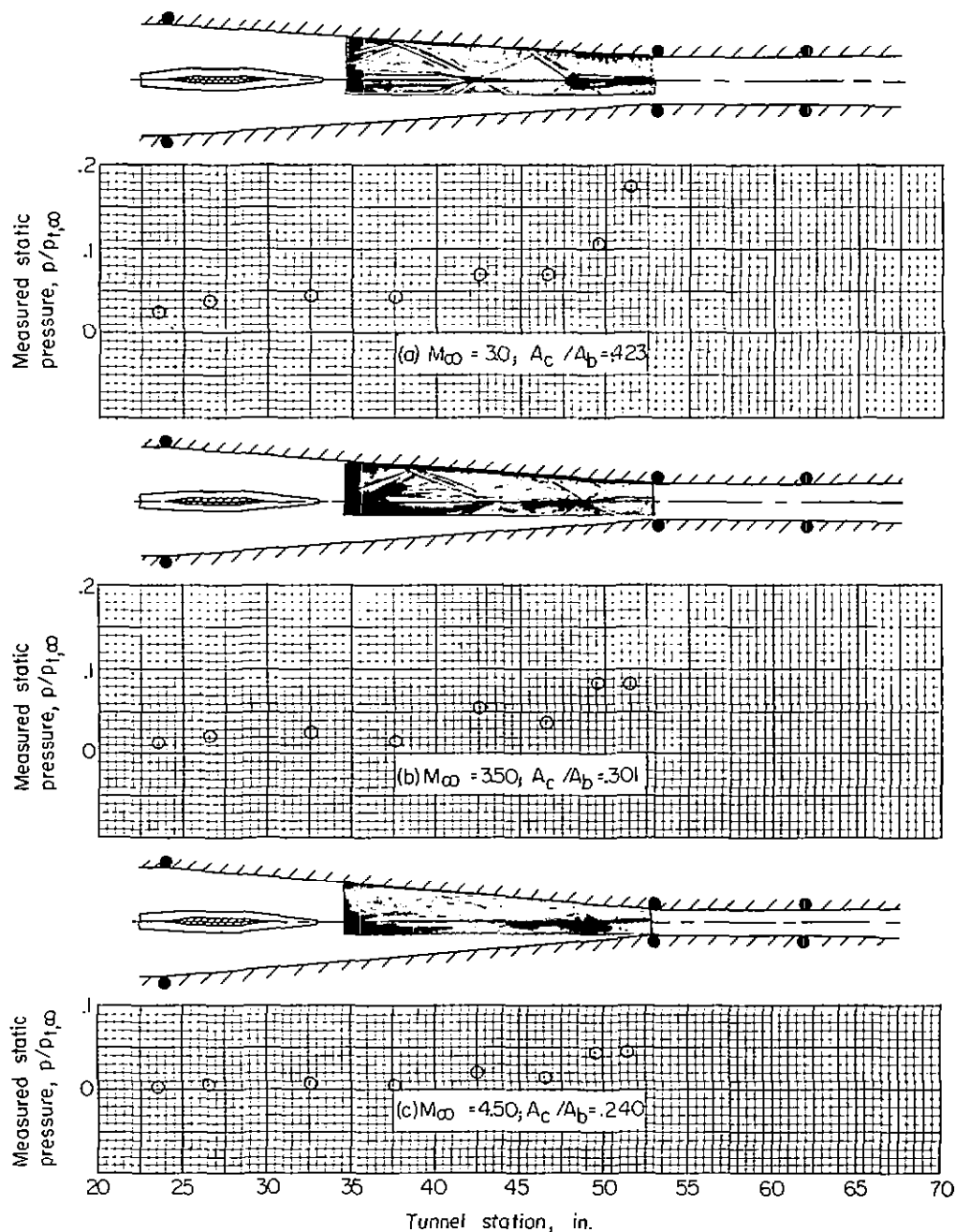
Figure 17.- Typical static pressure distributions on moveable wall and flow patterns in supersonic diffuser at near minimum values of minimum-running contraction ratio. With model; $p_{t,\infty}$ = 120 pounds per square inch absolute.
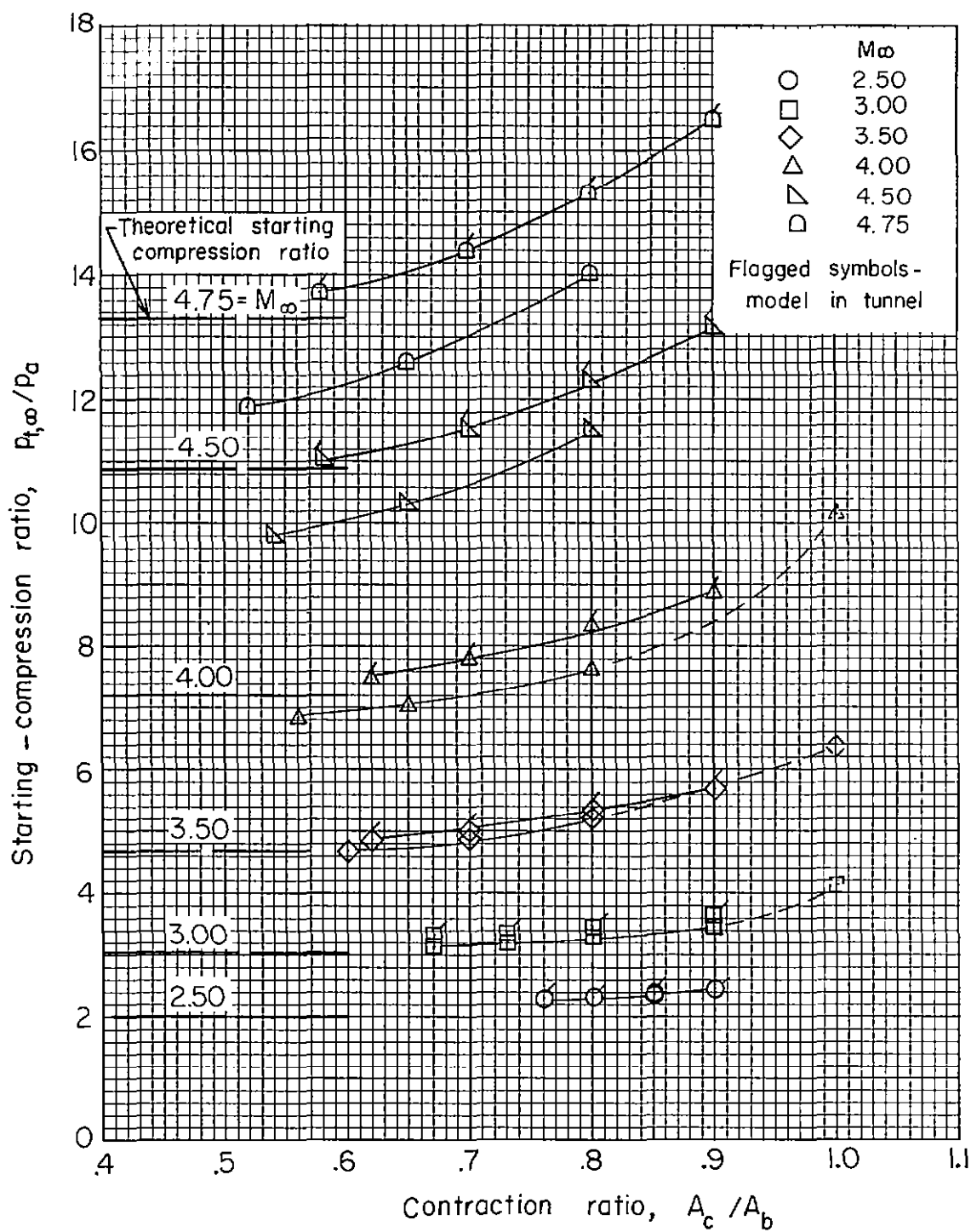
Figure 18.- Starting compression-ratio characteristics of diffuser
apparatus without injector.
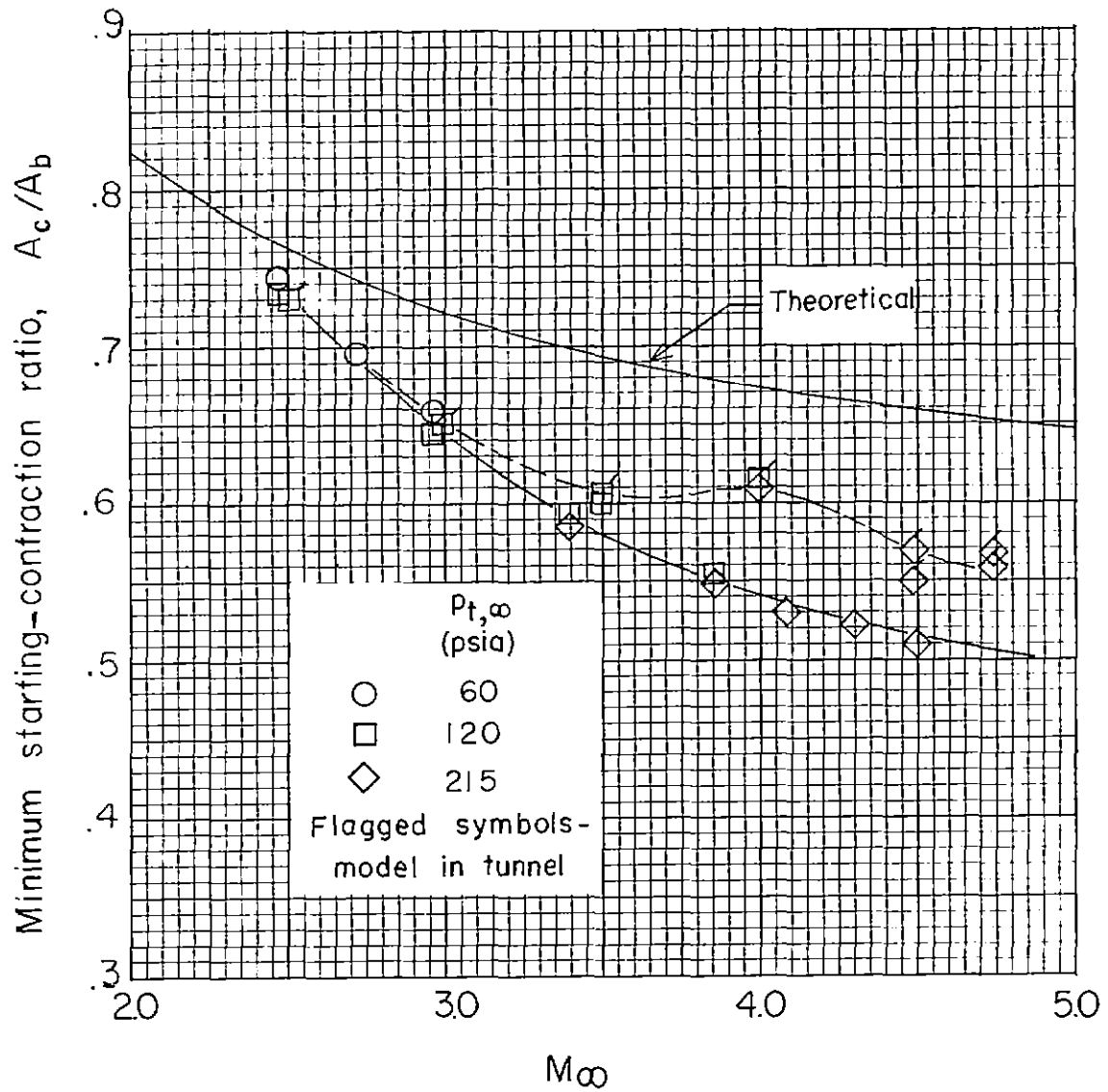
Figure 19.- Minimum starting-contraction ratio characteristics of dif-
fuser apparatus without injector.
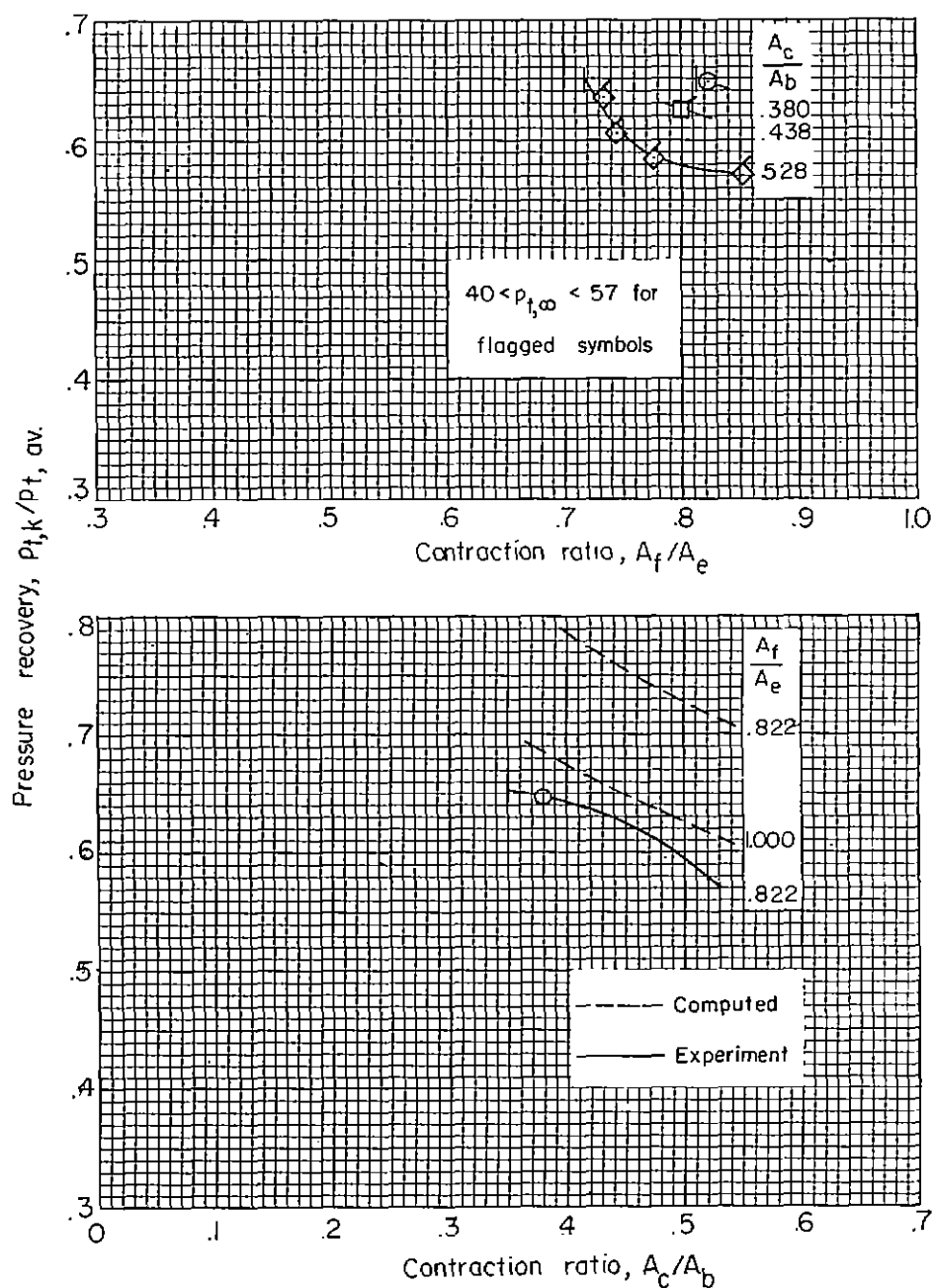
(a) M_∞ = 2.95.

Figure 20.- Pressure-recovery characteristics of diffuser apparatus
with injector and without model in tunnel. $p_{t,\infty}$ = 120 pounds per
square inch absolute unless noted.
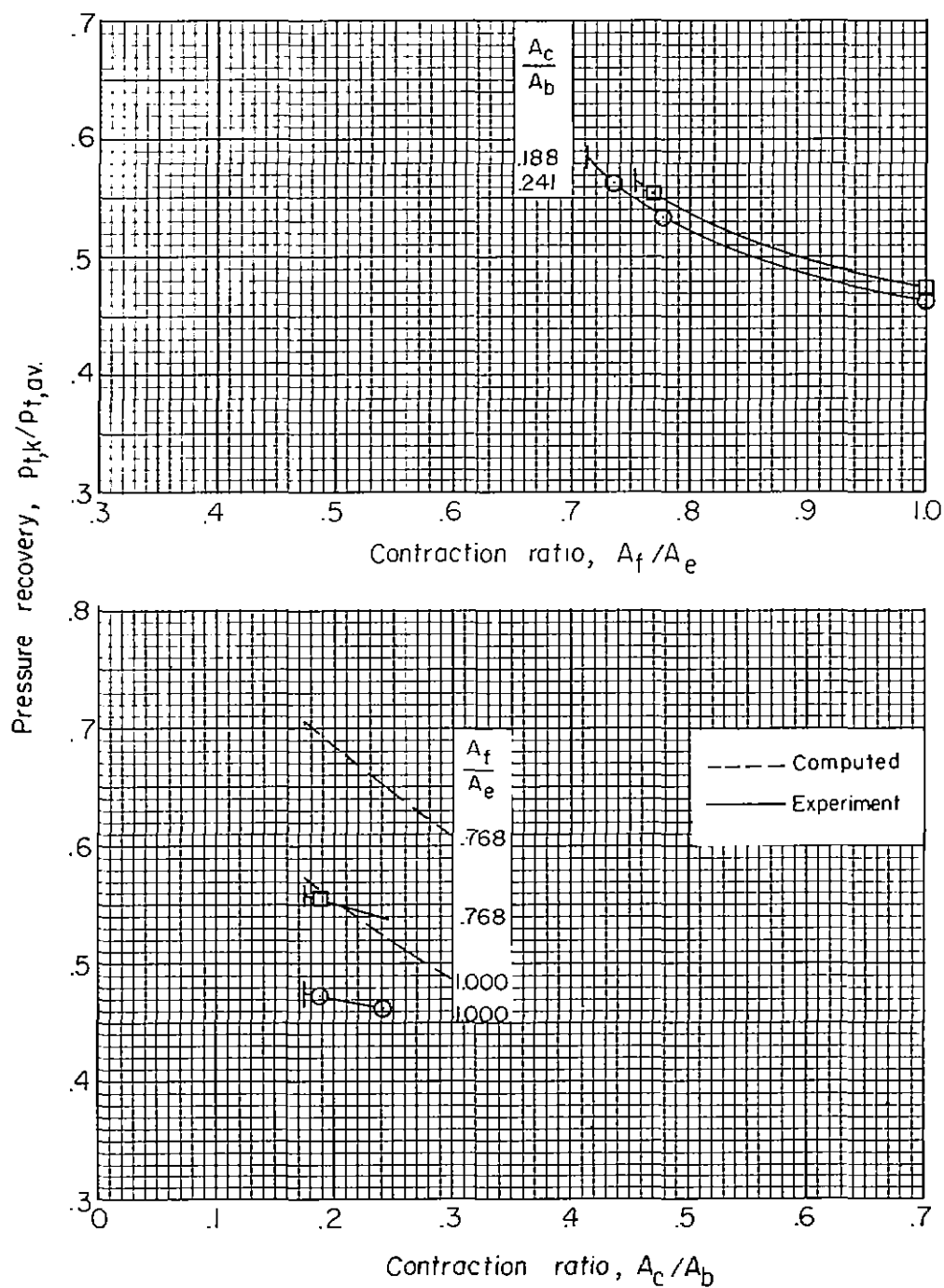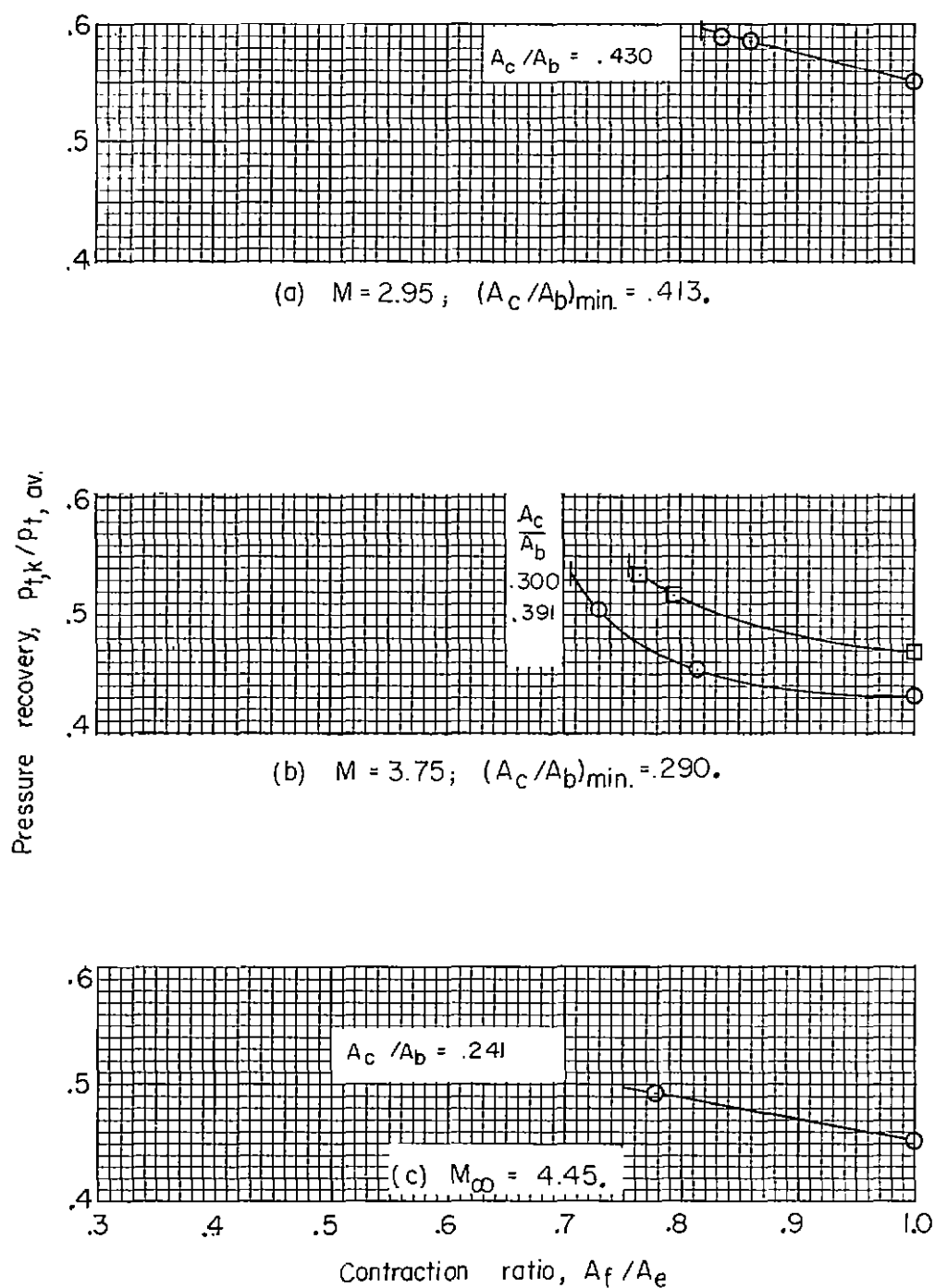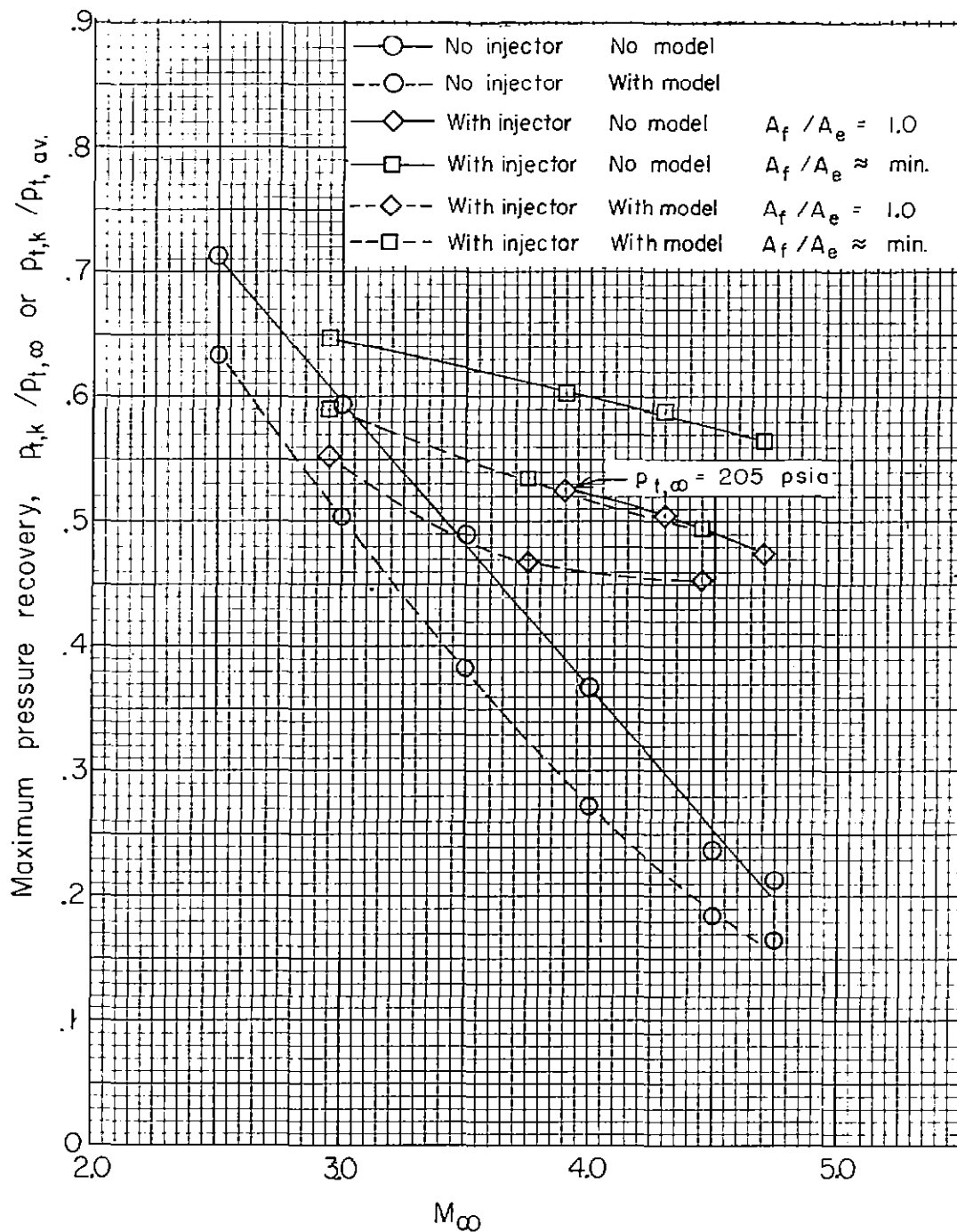
(b) $M_\infty = 3.90$.

Figure 20.- Continued.
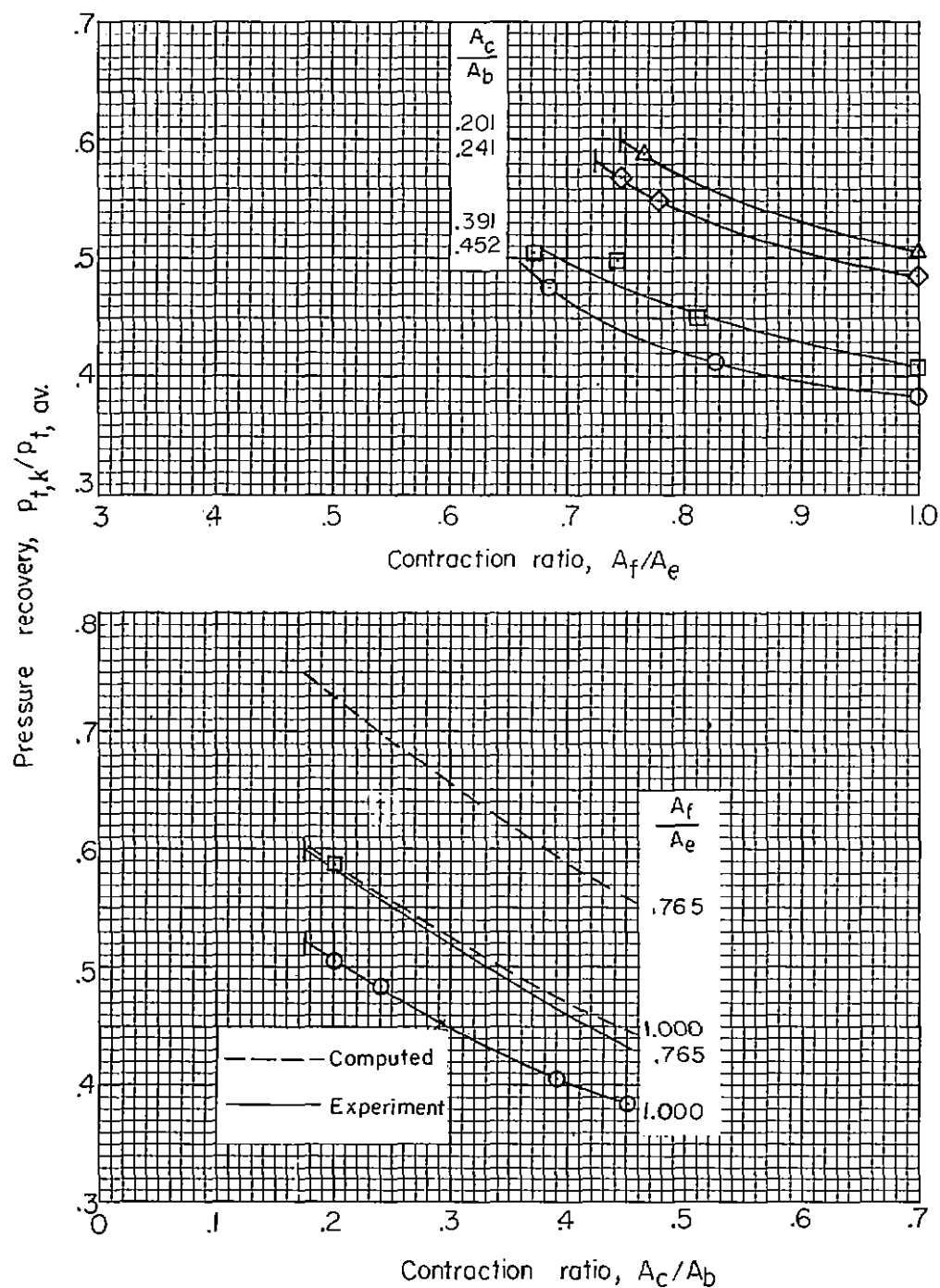
(c) $M_\infty = 4.30.$

Figure 20.- Continued.

(d) $M_\infty = 4.70$.

Figure 20.- Concluded.

(a) $M = 2.95$; $(A_c/A_b)_{min.} = .413$.

(b) $M = 3.75$; $(A_c/A_b)_{min.} = .290$.

(c) $M_\infty = 4.45$.

Figure 21.- Pressure-recovery characteristics of diffuser apparatus with injector and with model in tunnel. $p_{t,\infty} = 120$ pounds per square inch absolute.
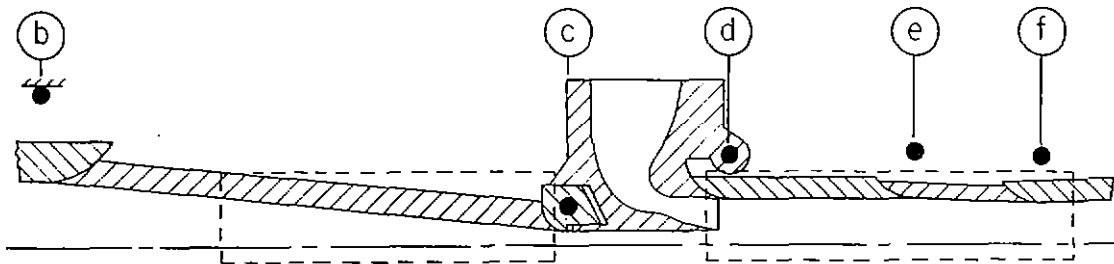
Figure 22.- Comparison of maximum pressure-recovery characteristics of diffuser apparatus with and without injector. $p_{t,\infty}$ = 120 pounds per square inch absolute unless noted.

(a) $M_\infty = 2.95$; $A_c/A_b = .379$; $A_f/A_e = .822$.

(b) $M_\infty = 3.90$; $A_c/A_b = .240$; $A_f/A_e = .772$.

(c) $M_\infty = 4.30$; $A_c/A_e = .202$; $A_f/A_e = .765$.

(d) $M_\infty = 4.70$; $A_c/A_b = .187$; $A_f/A_e = .760$.

(e) $M_\infty = 4.70$; $A_c/A_b = .240$; $A_f/A_e = .734$.

(f) $M_\infty = 4.70$; $A_c/A_b = .259$; $A_f/A_e = .970$.

(g) $M_\infty = 4.70$; $A_c/A_b = .270$; $A_f/A_e = .970$.

L-57-2729
Figure 23.- Typical shadowgraph of flow in diffuser apparatus with
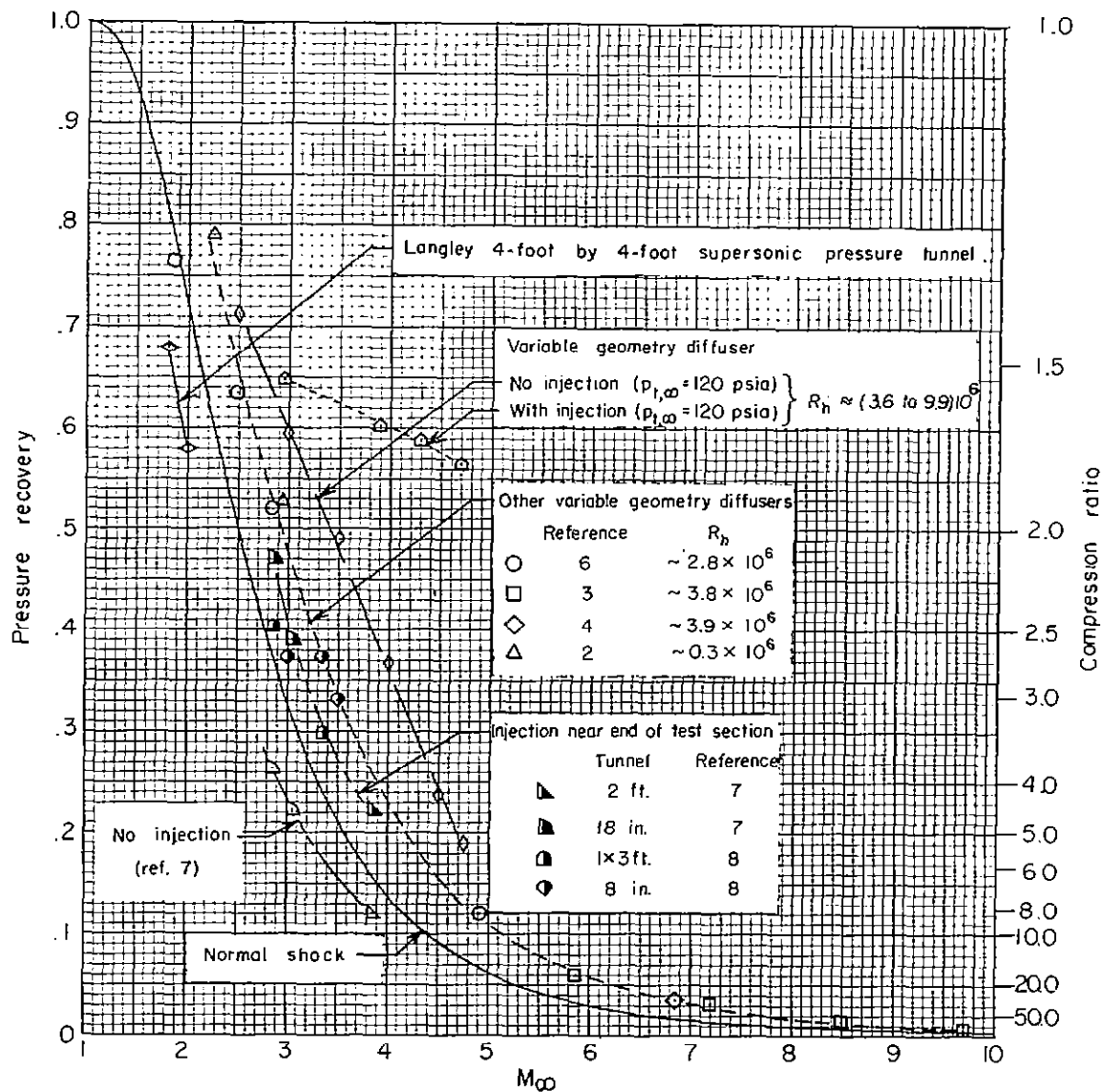injector installed. No model in airstream.

Figure 24.- Pressure-recovery characteristics of various wind-tunnel diffuser configurations.
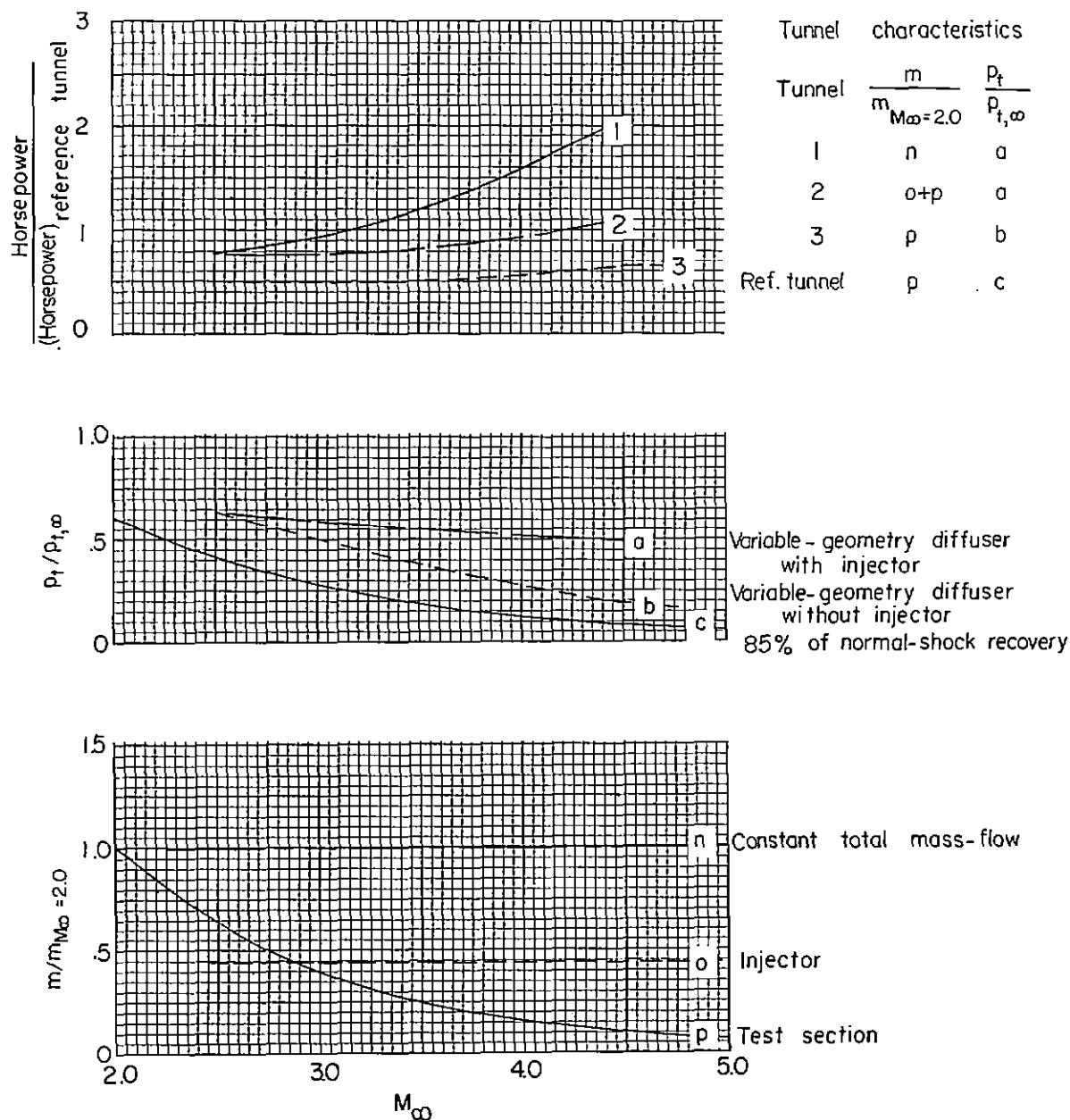
Figure 25.- Horsepower characteristics of several hypothetical wind
    tunnels, and pressure-recovery and mass-flow characteristics of
    these tunnels.